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27 DECEMBER 1976

EARTH VIEWING APPLICATIONS LABORATORY (EVAL)

FINAL REPORT

DEDICATED PAYLOAD, STANDARD TEST RACK PAYLOAD, SENSOR MODIFICATIONS

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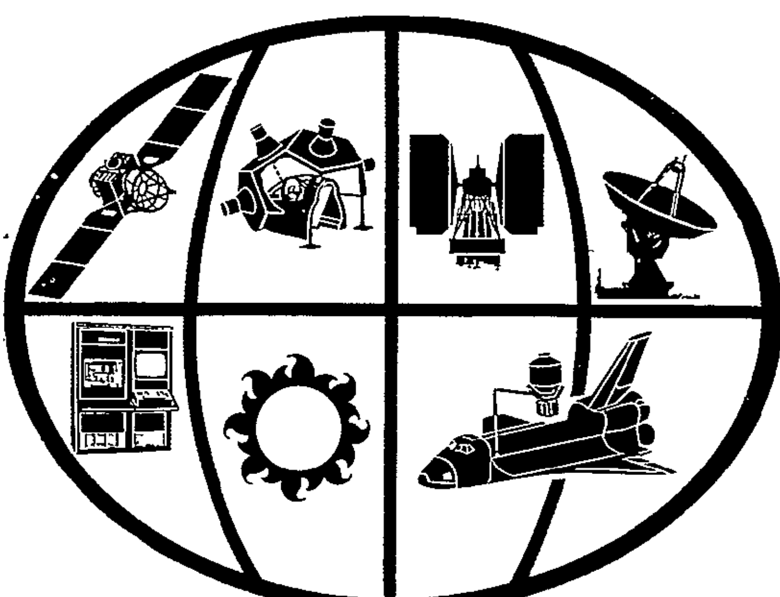

space division

CONTRACT NAS 5-23412

MODIFICATION NO. 39

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GENERAL ELECTRIC
SPACE DIVISION
Valley Forge Space Center
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PREFACE

The experiments and missions* described in this report, and the primary sensors required for their successful accomplishment, have been extracted from a recommended list prepared by the EVAL Steering Committee and discipline Working Groups. These earth-viewing experiments generally embody the characteristics of near term monetary value and human impact; and the required sensors have been judged to be available for a 1982 flight. The actual selection and mix of the experiments and sensors from this list was performed under the guidelines of creating a cost-effective payload.

The EVAL Steering Committee is comprised of the following individuals:

D. McConnell	NASA Headquarters	Chairman
H. Plotkin	NASA GSFC	Executive Secretary and Study Scientist
F. Flatow	NASA GSFC	Study Manager
J. Raper	NASA LARC	Environmental Quality
C. Laughlin	NASA GSFC	Weather and Climate
R. Moke	NASA JSC	Earth Resources
J. McGoogan	NASA WFC	Earth and Ocean Dynamics
E. Wolff	NASA GSFC	Communication and Navigation

*The terms "experiment" and "mission" are used somewhat interchangeably within this report to describe the various applications associated with this payload. In general, the distinction is on the degree of operationality of the application - those applications performing an operational function or end-to-end systems test are considered missions; while applications involved with sensor or technique development are identified as experiments.

ACKNOWLEDGEMENTS

Appreciation is expressed to the team of NASA/MSFC individuals led by R. Valentine and R. Davies for their contribution to the Earth Viewing Shuttle/Spacelab payloads described within this report. Inputs in the areas of cloud cover and the companion free flying satellite payloads associated with the EVAL STR payload have significantly enhanced the content of this study.

Previously published reports relating to this study include the following:

- EVAL Mission Requirements, General Electric, 76SDS4227, 7 May 1976
- Space Shuttle Earth Observation Sensors Pointing and Stability Requirements Study, General Electric, 76SDS4228, 7 May 1976
- Earth Viewing Applications Laboratory Instrument Catalog, General Electric, 25 May 1976
- EVAL System Concept Definition/Partial Spacelab Payload, General Electric, 76SDS4269, 30 September 1976 (Technical Report and Appendices)

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ACRONYMS AND ABBREVIATIONS

ALT	-	Altimeter (GEOS-C)
AMPA	-	Adaptive Multibeam Phased Array
ATC	-	Active Thermal Control
C&DMS	-	Command and Data Management System
CIMATS	-	Correlation Interferometer for the Measurement of Atmospheric Trace Species
CONUS	-	Continental United States
CPR	-	Cloud Physics Radiometer
ECS	-	Environmental Control System
EMI	-	Electromagnetic Interference
EPDS	-	Electrical Power and Distribution System
ETR	-	Eastern Test Range
EVAL	-	Earth Viewing Applications Laboratory
FOV	-	Field of View
GEOS	-	Geodetic Satellite
GSFC	-	Goddard Space Flight Center
HIRS	-	High Resolution Infrared Radiometer System
HRDR	-	High Rate Data Recorder
HRM	-	High Rate Multiplexer
IMU	-	Inertial Measurement Unit
I/O	-	Input/Output
IS	-	Interface Station
IUS	-	Interim Upper Stage
JSC	-	Johnson Space Center
LARC	-	Langley Research Center
LFC	-	Large Format Camera
LRS	-	Laser Ranging System
MAPS	-	Measurement of Air Pollution From Satellites
MM	-	Miniaturized Pointing Mount (Minimount)
MSFC	-	Marshall Space Flight Center
NASA	-	National Aeronautics and Space Agency
OEDSF	-	Onboard Experiment Data Support Facility
PSA	-	Post Sleep Activities
RAU	-	Remote Acquisition Unit
RCS	-	Reaction Control System
SAR	-	Synthetic Aperture Radar
SBUV	-	Solar Backscatter Ultraviolet
SIMS	-	Shuttle Imaging Microwave System
SIR	-	Shuttle Imaging Radar
SMM	-	Solar Max Mission
STDN	-	Spacecraft Tracking and Data Network
STR	-	Standard Test Rack
STS	-	Shuttle Transportation System
TDRS(S)	-	Tracking and Data Relay Satellite (System)
TM	-	Thematic Mapper
TOMS	-	Total Ozone Mapping Spectrometer
VHDRR	-	Very High Data Rate Recorder
WFC	-	Wallops Flight Center

SECTION 1
INTRODUCTION

This report extends the preliminary analysis of strawman earth-viewing Shuttle sortie payloads begun with the partial Spacelab payload analyzed in GE report 76SDS4269, dated 30 September 1976. The payloads analyzed in this report essentially represent the two extremes of Shuttle sortie application payloads: a full Shuttle sortie payload dedicated to earth-viewing applications, and a small structure payload which can fly on a space available basis with another primary Shuttle payload such as a free flying satellite. The intent of the dedicated mission analysis was to configure an ambitious, but feasible, payload; which, while rich in scientific return, would also stress the system and reveal any deficiencies or problem areas in mission planning, support equipment, and operations. Conversely, the intent of the small structure payload was to demonstrate the ease with which a small, simple, flexible payload can be accommodated on Shuttle flights. Analyses of these payloads are presented in Sections 2 and 3 of this report.

The final section of this report, Section 4, is devoted to a preliminary analysis of sensor modifications required for those sensors associated with the various strawman payloads analyzed to date under EVAL. (Sensors associated with the partial Spacelab payload previously mentioned are included along with those for the dedicated full Shuttle and small structure payloads described in this report). While some of the sensors associated with the EVAL payloads are new developments being specifically designed for use on Shuttle, the majority of sensors considered have been/are being developed for other platforms such as satellites, sounding rockets, balloons, or aircraft. Generally, these sensors require modification to be compatible with the Shuttle environment and orbital conditions.

SECTION 2

EVAL DEDICATED SHUTTLE PAYLOAD

This section addresses the preliminary design of a full Shuttle sortie payload dedicated to earth-viewing applications. Mission parameters associated with this flight include a launch date of August 1982, an inclination of 57°, and an orbital altitude of 200 km. The guideline for mission duration was to remain in orbit as long as Shuttle/Spacelab resources would permit.

The basic payload carrier associated with this flight consists of the Spacelab configuration defined as the long module plus pallet(s), complemented by a STR (Standard Test Rack).* The Spacelab configuration is shown in Figure 2-1. This figure shows a long module plus two pallets; however, as will be described in the subsequent payload description section for this payload, a self-contained sensor, the Shuttle Imaging Microwave System (SIMS), having its own pallet like structure is substituted for the forward pallet (closest to the module).

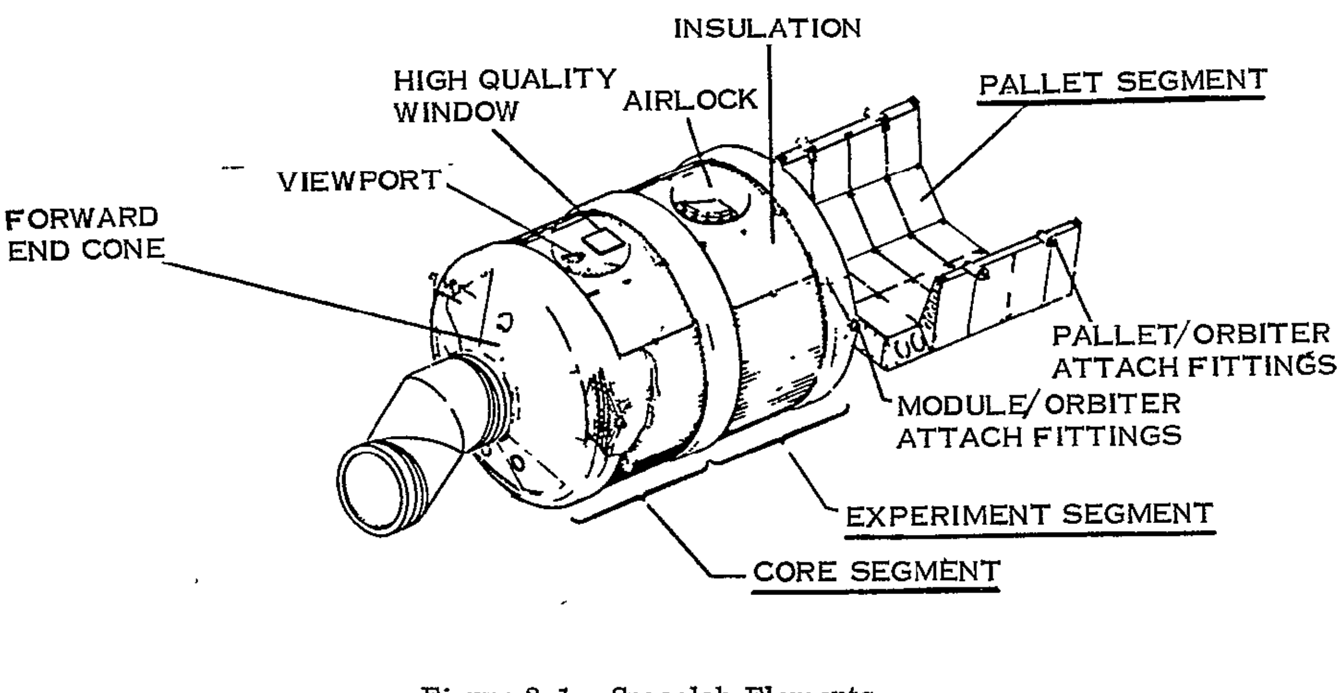


Figure 2-1. Spacelab Elements

* The basis for the Standard Test Rack is the Standard Earth Observations Package for Shuttle (SEOPS) which has been conceptually developed by General Electric under contract to NASA/JSC.

A STR bridge configuration, pictured in Figure 2-2, is used around the Spacelab transfer tunnel. The STR is a modular system of structures and subsystems which accommodates various sensors and interfaces with Shuttle in a nearly autonomous manner. The combined Spacelab plus STR configuration considered for this payload is illustrated in Figure 2-3.

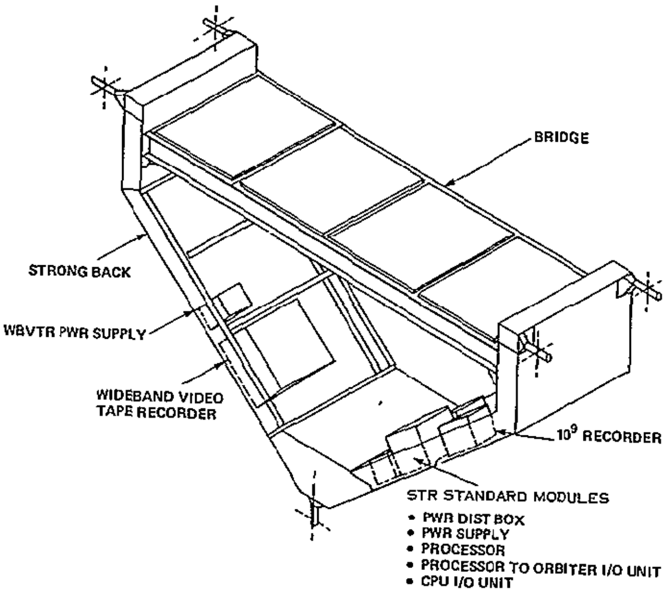


Figure 2-2. STR Bridge Configuration

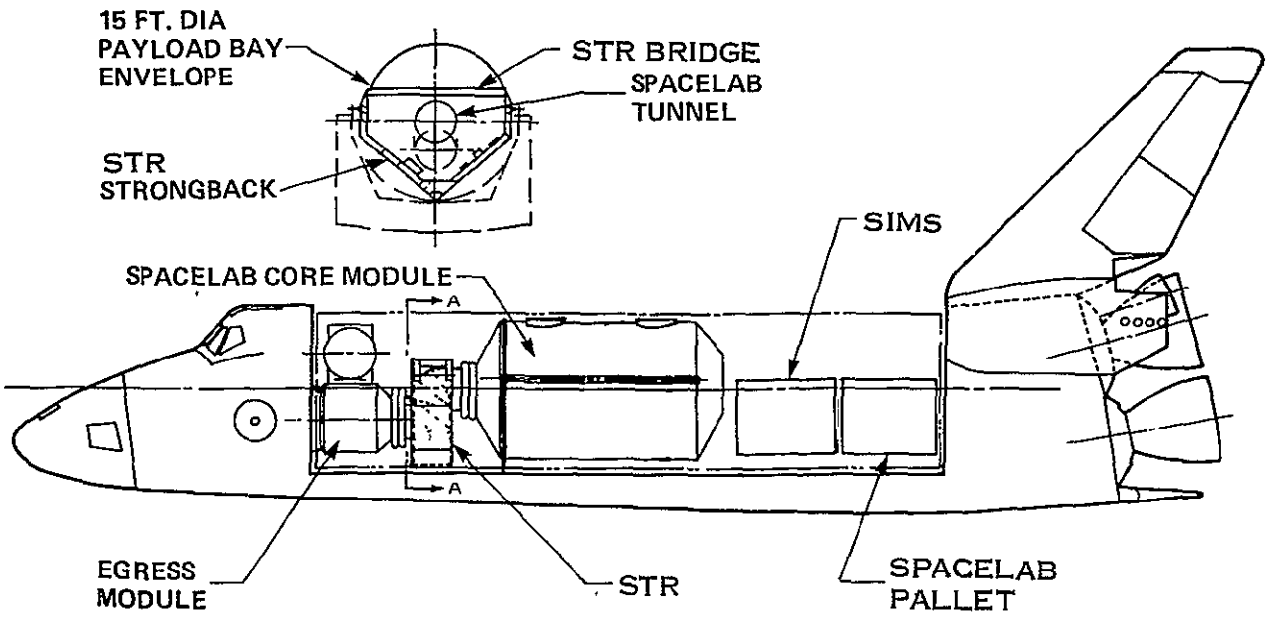


Figure 2-3. Typical STR Bridge Installation with Spacelab

2.1 ACCOMMODATIONS

Accommodations for EVAL experiments will be provided by elements of Spacelab, STR, and, to some extent, the Shuttle Orbiter. Details of the pertinent capabilities provided by these systems are described in the following paragraphs.

2.1.1 ORBITER

From an experimental standpoint the Orbiter provides orbital position and location, gross pointing and attitude control, and crew support.

Orbital Position Determination

Knowledge of the orbital position of the Orbiter/Spacelab/experiment at any time is dependent on the elapsed time since the last tracking pass and the tracking system used. The on-orbit navigation accuracies, using the Spacecraft Tracking and Data Network (STDN) and the Tracking and Data Relay Satellite (TDRS) system are given in Table 2-1 for a 185 km (100 nm) altitude case. (This is the only information presently available). These expected accuracies will obviously be somewhat degraded for the 200 km (108 nm) orbit considered for this payload.

Pointing and Attitude Control

The Shuttle Orbiter has the capability of achieving and maintaining any desired space or earth referenced attitude with respect to either the Orbiter navigation base or a payload provided and mounted sensor. The pointing accuracy, however, is a function of the error sources associated with the characteristics of the particular attitude sensor, the type of control system, and the Orbiter flexure.

Table 2-1. Expected On-Orbit Navigation Accuracies (3 Sigma) for 100 Nautical Miles (185 km) Orbital Altitude

Navigation System	Position, Feet (Meters)				Velocity, Feet/Sec (Meters/Sec)			
	Altitude	Down-track	Cross-track	Root Sum Square	Altitude	Down-track	Cross-track	Root Sum Square
STDN								
After last tracking pass	440 (130)	370 (110)	430 (130)	730 (220)	3.9 (1.2)	0.5 (0.15)	2.0 (0.6)	4.4 (1.3)
After one revolution	470 (150)	850 (260)	430 (130)	1030 (315)	4.3 (1.3)	0.5 (0.15)	2.0 (0.6)	4.8 (1.4)
TDRS								
After last tracking pass	300 (90)	1400 (430)	1520 (460)	2070 (630)	1.6 (0.5)	0.35 (0.11)	0.5 (0.15)	1.7 (0.5)
After one revolution	300 (90)	2010 (610)	1520 (460)	2400 (740)	2.4 (0.7)	0.3 (0.1)	0.5 (0.15)	2.5 (0.7)

The Orbiter Inertial Measurement Unit (IMU), located in the Orbiter cabin, is used to supply inertial attitude reference signals; and, in conjunction with the onboard navigation system, can provide a pointing capability of the navigation base accurate to within $\pm 0.5^{\circ}$ for earth-viewing missions. This pointing accuracy can degrade to approximately $\pm 2.0^{\circ}$ for payloads located in the aft bay due to structural flexure of the Shuttle vehicle, payload structural and mounting misalignments, and calibration errors with respect to the navigation base. In order to provide greater accuracy in payload pointing, the Orbiter is capable of accepting error signals from a more accurate payload supplied and mounted sensor. In this case, the Orbiter is capable of maintaining a specified attitude to within ± 0.1 deg/axis by using the full capability of the Reaction Control System (RCS) jets, and a stability rate of ± 0.01 deg/sec/axis.

Crew Support

The Orbiter consists of the commander and pilot to operate and manage the Orbiter, a mission specialist, and one or more payload specialists. While both the commander and pilot will be primarily occupied with operating the Orbiter, they may support/perform specific payload operations if appropriate, and at the discretion of the individual experiment sponsors. The mission specialist will be responsible for the coordination of overall Orbiter operations in the areas of flight planning, consumable usage and other activities affecting payload operations. At the discretion of the individual experiment sponsors he may also assist in the experiment operations, and may in specific cases serve as the payload specialist. The payload specialist(s) will be responsible for the attainment of experiment objectives (this individual may be the actual experimenter or a designated representative); including the operation of experiment equipment. Up to four payload specialists can be accommodated.

2.1.2 SPACELAB

Spacelab, as utilized by this payload, consists of two basic elements - a pressurized module and an unpressurized pallet. The module provides a controlled, pressurized environment for the users and their equipment, and supplies basic services such as power, thermal control, and data management together with certain basic support equipment such as standard racks, scientific airlocks, etc., which may be used as required. The pallet is an unpressurized platform to which instruments such as cameras and antennas that require direct exposure to space may be mounted. The pallet provides some basic services, such as power conditioning and distribution, data distribution, and thermal control.

Pressurized Module

The module is a cylindrical pressure shell measuring 4060 mm in diameter and 6964.6 mm in length. It contains subsystem equipment for Spacelab, crew work space, rack volume for experiment installation, and an optical window and an airlock on the top for mounting small instruments which may require manned operation. Figure 2-4 depicts cutaway sections of the pressurized module. 22.2 m^3 of space is available for experiment equipment, including all space and ceiling storage compartment.

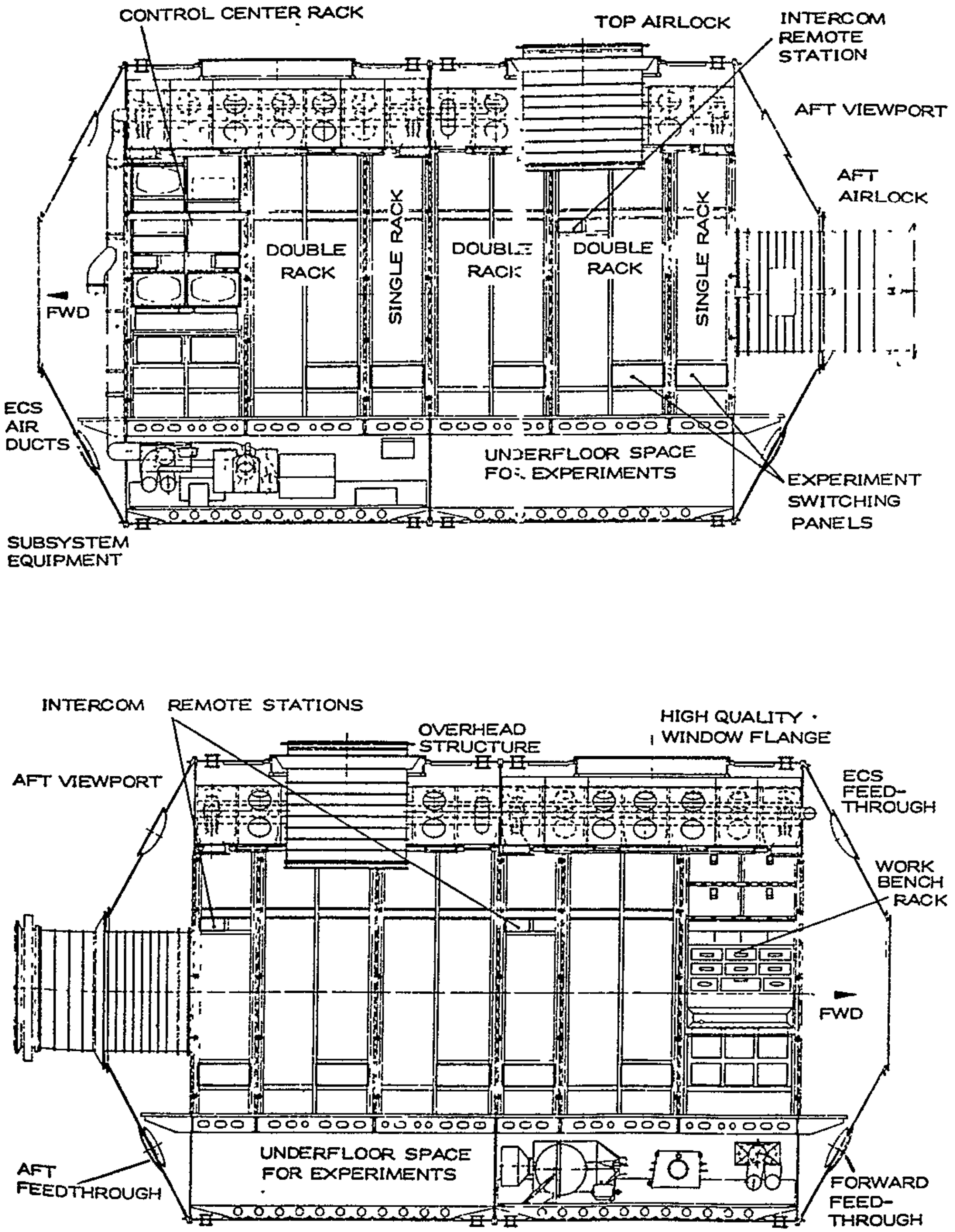


Figure 2-4. Pressurized Module Cutaway

Pallet

Basically, the Spacelab pallet is an unpressurized platform to which instruments that require direct exposure to space may be mounted. The U-shaped pallet, shown in Figure 2-5, is approximately 2.9 meters long and 4.0 meters in width; and provides basic services such as power conditioning and distribution, data distribution, and thermal control. The pallet structure for accommodating experiment equipment, Figure 2-6, provides mounting support for the experiments either directly on skin panels or through specific hardpoints for better dispersion of concentrated loads. The inner side and floor panels can support loads of 50 kg/m^2, whereas the outer panels can support 10 kg/m^2. If experiment equipment exceeds the panel load capability, it can be mounted only on standard equipment hard points. Provisions for 24 hard points are located on the inner surface at the intersection of the frames and longitudinal members, as shown in Figure 2-6. Each hard point provides a dynamic load-carrying capability of: Xp = 28,547N, Yp = 18,443N, and Zp = 75,046N. The overall payload carrying capability of the pallet is 1100 kg/m (uniformly distributed over the pallet) with a CG limitation between 25 mm above the pallet floor line and the Orbiter bay horizontal centerline. From an area and volume standpoint a single pallet provides approximately 17 m^2 of mounting area and 33 m^3 volume above the floor.

The Spacelab Electrical Power and Distribution Subsystem (EPDS) receives its primary power from the Orbiter: 7 kW average and 12 kW peak are delivered during orbital operations. The power available for experiments is the resultant after mission dependent and mission independent equipment power consumption is subtracted from that supplied by the Orbiter. For the long module plus single pallet Spacelab configuration used for this payload, maximums of 3.25 kW average and 7.25 kW peak exist for the payload. The total energy available to the payload is TBD* kWh. The power bus system running through the module and pallets provides the wiring for primary dc (28 Vdc nominal) and 115/200 Vac at 400 Hz. On the pallet, payload equipment is hardwired into the distribution bus. Figure 2-7 illustrates the power distribution scheme for the module plus pallet Spacelab configuration.

* An exact value is not presently available for this Spacelab configuration; however, based on interpolation of published data for other configurations (long module only - 420 kWh, short module plus three pallets - 369 kWh) it is estimated that approximately 400 kWh will be available for this payload.

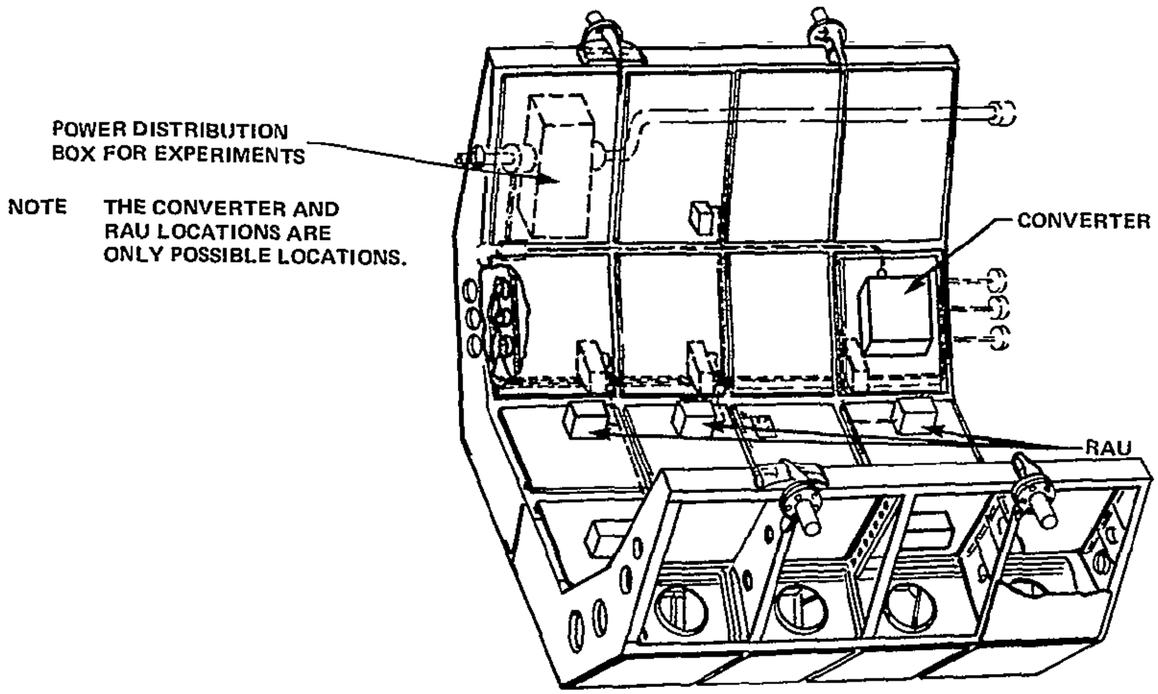


Figure 2-5. Spacelab Pallet

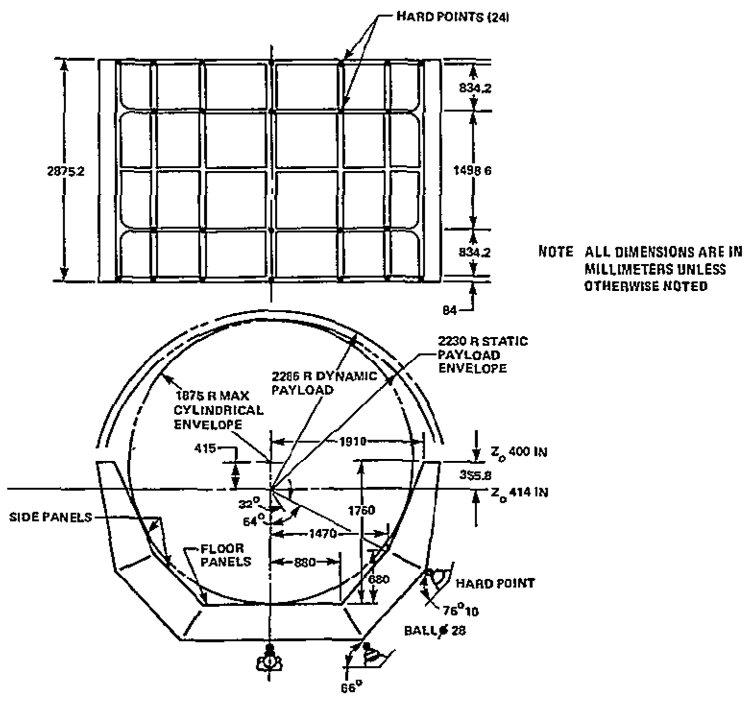


Figure 2-6. Pallet Structure

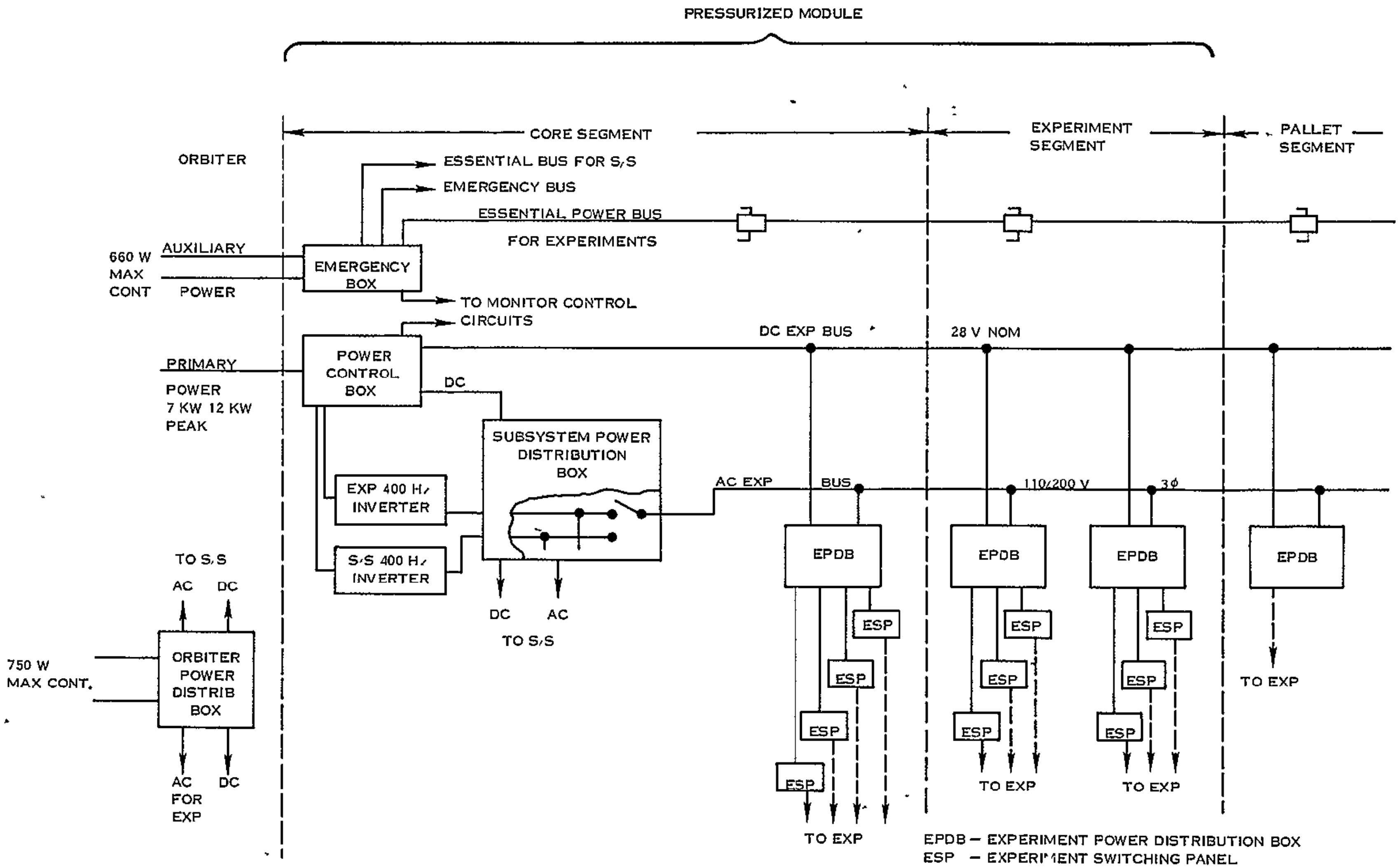


Figure 2-7. Power Distribution Scheme, Module/Pallet Configurations (Combines Mission Independent and Mission Dependent Equipment)

Environmental control for experiments on the pallet is provided by cooling loops and the use of cold plates and thermal capacitors. Eight cold plates (capability 24-40°C) and up to four thermal capacitors are available to dissipate peak heat loads. The maximum capability per cold plate is 1 kW. Figure 2-8 shows the characteristics and location of these devices. Air cooling loops control the module atmosphere between 18-27°C. Experiment racks are cooled (22-40°C) by the avionics air cooling loop and a liquid-to-liquid experiment heat exchanger.

Remote acquisition units (RAU's) are the principal interface between experiments and the command and data management subsystem. Up to four RAU's can be provided on the pallet. High frequency analog data is accommodated by an analog channel using a high rate multiplexer. Digital data can be stored by a recorder; however, the maximum data rate allowable is 30 Mbps. Up to 20 minutes of data storage can be accommodated at the 30 Mbps rate.

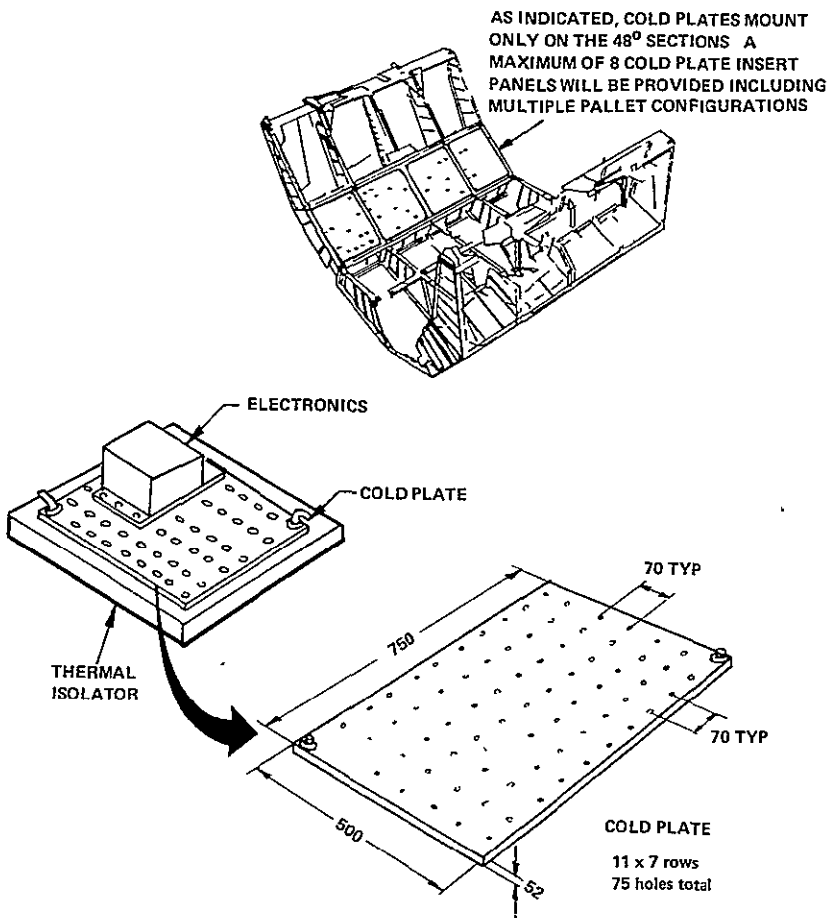


Figure 2-8. Cold Plate Mounting

The available field of view above the Spacelab pallet with the Orbiter cargo bay doors and radiators open is variable forward and aft dependent upon the Spacelab configuration and the location of the pallet. The field-of-view is restricted in these directions by either the Spacelab pressurized module or the Orbiter cabin and the Orbiter empennage. Figure 2-9 shows limiting examples for this situation. The side field-of-view limitations are constant as shown in Figure 2-10.

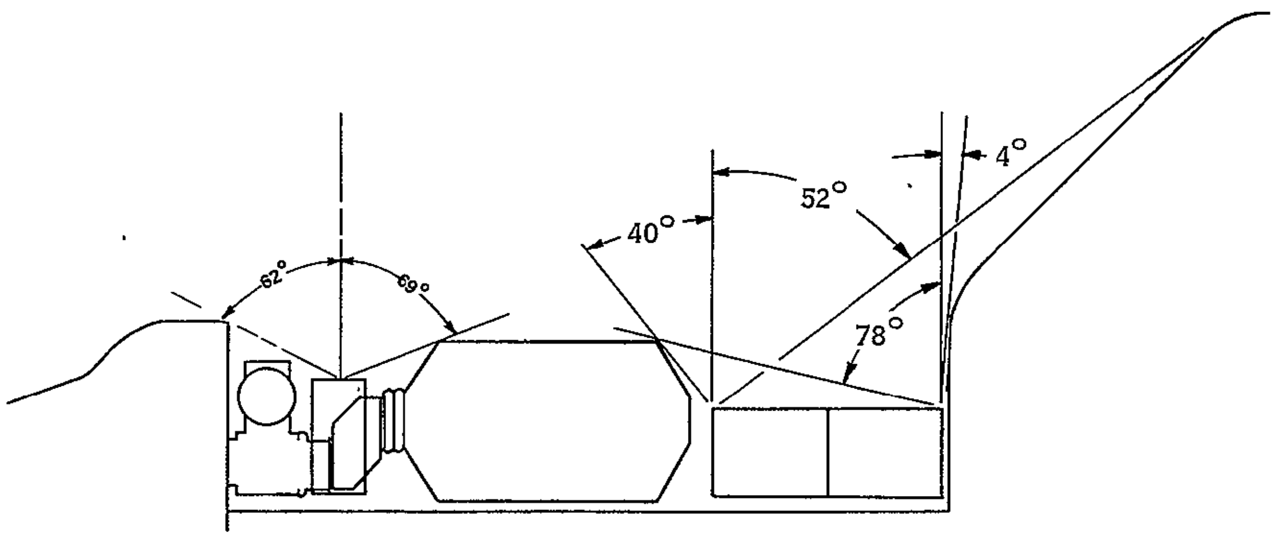


Figure 2-9. Limiting Fields of View

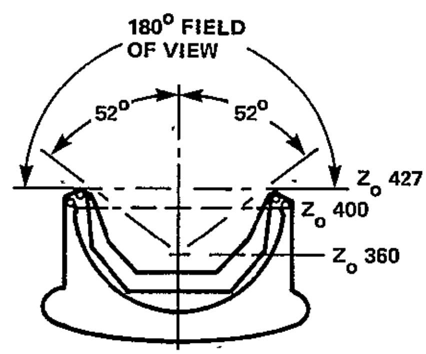


Figure 2-10. Orbiter Field of View-Side

2.1.3 STR.

The STR system shown in Figure 2-11, consists of a modular structure and support subsystems. Since STR is independent of Spacelab, its accommodations are somewhat unique.

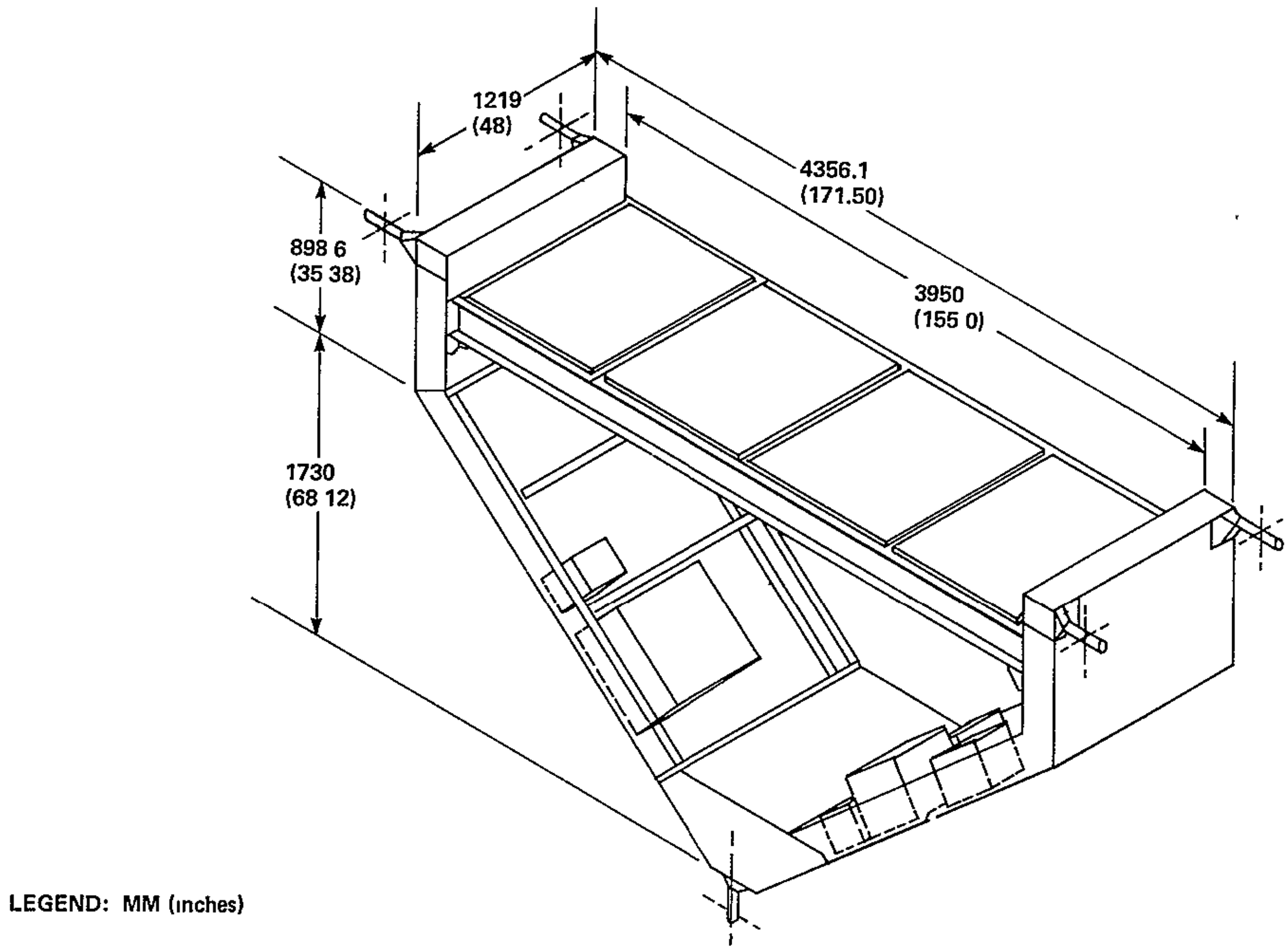


Figure 2-11. STR Bridge Configuration

The STR structure consists of a strongback, which provides the base for the bridge. The strongback is U-shaped, providing clearance around the Spacelab tunnel, and transmits the STR loads to the trunnion fittings at the keel and side attachment points. Generally the sensors are mounted on the bridge with the STR support subsystems attached to the strongback. The structural weight for the STR bridge configuration is 313 kg. This configuration can support a payload weight of 1043 kg. For earth viewing applications approximately 6 m^2 and 23.7 m^3 of mounting surface and volume are available.

The STR support subsystems provide alignment and rate knowledge, conditioned electrical power, temperature control, and data management and processing. STR basically depends on the Orbiter attitude control subsystem for target pointing and stability.

STR does provide alignment of the instruments with the Orbiter plane within 0.5° and utilizes a self-contained star tracker to provide attitude update for pitch, roll and yaw. Residual rate knowledge to 0.0001° per second is provided via a gyro package.

STR uses electrical energy from the Shuttle Orbiter main DC-2 bus, regulates it, and distributes it to the attached sensors and electronic boxes. Maximum power availability with this system is 3 kW at +28 Vdc $\pm 2\%$.

Thermal control is maintained within $\pm$ $8^{\circ}C$ between $5^{\circ}C$ and $21^{\circ}C$ using a passive and louver system. STR can provide its own data handling, processing, and storage. Specific functions performed by this subsystem include sensor and subsystem command generation, housekeeping data formatting and processing, system checkout and evaluation, sensor data processing, recording and transmission control, and signal routing. The STR can be reprogrammed from the ground, or it can transmit data to the ground through the Orbiter command and data management system. STR capabilities include command and telemetry provided by the modular addition of hardware and firmware circuits capable of handling up to 240 mbps, and two types of tape recorders: a 240 mb wideband tape recorder and a NASA standard narrowband (10^8, 10^9) tape recorder.

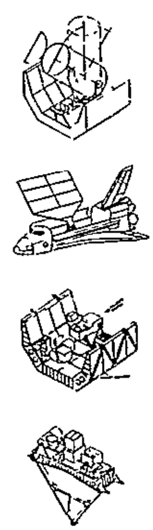
Orbiter data available to STR payloads include ephemeris, time, attitude, and caution/warning.

2.2 PAYLOAD DESCRIPTION

The payload specified for this flight is a multi-discipline grouping of experiments selected from a collection of high priority experiments designated by the EVAL Steering Group and the individual discipline Working Groups as being available for a 1982 Shuttle flight. Experiments are included in this payload representing the discipline of Earth Resources, Weather and Climate, Earth and Ocean Dynamics, and Communication and Navigation. The emphasis of this payload, however, is on the discipline of Earth Resource and Earth and Ocean Dynamics. This emphasis is due to a combination of factors: (1) the low orbital altitude translates into high resolution capability, and repeated looks at the target areas which are desirable for these disciplines; and (2) many of the missions/experiments within these disciplines require multiple, large sensors which essentially necessitates a dedicated Shuttle payload.

The selection of specific experiments was based on maximizing benefits while minimizing costs. Commonality of equipment and synergistic enhancement of experiments thus were important factors in selecting the payload. The experiments selected for this payload accomplish one or more of the following roles: technique development, sensor development, application development, operational platform. A brief exploration of these roles is provided in Figure 2-12.

TECHNIQUE DEVELOPMENT	EARLY INVESTIGATIONS OF UNDERLYING SCIENTIFIC FRAMEWORK – SIGNATURES – CUT AND TRY – LAB INSTRUMENTS
SENSOR DEVELOPMENT	ENGINEERING DEVELOPMENT AND EXPERIMENTATION TO FINALIZE SENSOR DESIGN – PERFORMANCE VERIFICATION/CAL – INCREMENTAL BUILDUP
APPLICATION DEVELOPMENT	EXERCISING OF A PROTOTYPE END TO END APPLICATIONS SYSTEM TO DEMONSTRATE OPERATIONAL POTENTIAL
OPERATIONAL PLATFORM	APPLICATIONS ROUTINELY CARRIED OUT TO SATISFY INFORMATION NEEDS OF AN OPERATIONAL RESOURCE MANAGER



EACH ROLE PARALLELS A STEP IN APPLICATIONS DEVELOPMENT

Figure 2-12. The Four Roles for Sortie Flights

2.2.1 MISSIONS

The EVAL experiments/missions selected for this payload are:

- Earth Resources

– Crop Survey	(Technique Development)
– Vegetation Stress	(Technique Development)
– Urban Planning	(Application Development, Operational Platform)
– Timber Inventory	(Application Development, Operational Platform)
– Range Condition Assessment	(Application Development)
– Mineral Exploration	(Application Development)
– Marine Resources	(Technique Development)
– Water Inventory	(Technique Development)

- Earth and Ocean Dynamics

– Crustal Motions	(Sensor Development, Technique Development)
– Sea Surface Temperature	(Applications Development)
– Ocean Currents	(Technique Development)
– Geoid Measurement	(Operational Platform)
– Sea Ice Assessment	(Applications Development)
– Storm Assessment	(Technique Development)

- Weather and Climate

– Cloud Climatology	(Sensor Development, Applications Development)
– Ozone Mapping	(Applications Development)

- Communications and Navigation

– Multibeam Communications	(Sensor Development, Applications Development)

A brief description of these experiments is provided in the following paragraphs. For a more detailed exploration the reader is referred to "EVAL Mission Requirements", 76SDS4227, General Electric Co., 7 May 1976, developed under NASA Contract No. NAS5-24022.

Timber Inventory. This mission will investigate the feasibility of surveying and monitoring forest lands to develop forecasts of timber production, productive status, and efficiency and ecological soundness of timber production and harvesting operations. Instruments required for this mission include a thematic mapper, a high resolution large format camera, and a Synthetic Aperture Radar.

Urban and Regional Planning. This mission will verify the use of remotely sensed data for inventorying land use to support the preparation of legally required comprehensive plans by urban and regional planners. Instruments required include a thematic mapper and a high resolution, large format camera.

Mineral Survey. This mission will investigate the use of remotely sensed data for detection of surface indicators of mineral deposits. Instruments required for this experiment include a thematic mapper, a high resolution stereo camera and a Synthetic Aperture Radar.

World Crop Survey. This experiment will investigate the feasibility of using combinations of remotely sensed data to periodically survey crops on a global basis in order to inventory acreage, predict yield and forecast world production. Sensors required for this mission include a thematic mapper, an imaging microwave radiometer system, a Synthetic Aperture Radar and a high resolution camera.

Vegetation Stress Detection. This experiment will investigate the feasibility of using combinations of remotely sensed data to detect and monitor major vegetation stress due to insect infestation, disease, flooding etc. in U. S. pasture and cropland. Sensors required for this mission include a high-resolution camera, a thematic mapper, a Synthetic Aperture Radar, and an imaging microwave radiometer system.

Range Condition Assessment. The purpose of this mission is to investigate the feasibility of using remotely sensed data to survey pasture and range areas, to prepare statistical summaries of forage acreages, to calculate supportive capacity for livestock, and to assess current grazing practices. The required instruments include a thematic mapper, a Synthetic Aperture Radar, and a Shuttle Imaging Microwave Radiometer.

Water Availability Forecasting. This experiment will investigate the use of remotely sensed data to provide forecasts of water availability for irrigation, hydroelectric power generation and shale cracking based on snow and soil moisture and appropriate runoff-prediction models. Required instruments include a thematic mapper, a Synthetic Apertur Radar and a Shuttle Imaging Microwave Radiometer.

Living Marine Resources Assessment Development Program. This experiment will inves tigate the use of remotely sensed data in specifying and monitoring the relationships between marine (environmental and biological) parameters and the habits and characteristic of living marine resources. Sensors required for this mission include a thematic mapper a Shuttle Imaging Microwave Radiometer, and a Synthetic Aperture Radar.

Crustal Motions. The purpose of this experiment is to test and demonstrate the applicatio of a precision spaceborne laser ranging system for measuring small relative crustal moti These results would be used in developing an operational system for detecting land subsidence and earthquake prediction. The laser ranging system is the primary sensor for this experiment, however the use of a large format camera and a multi-spectral scanner is also desirable.

Sea Surface Temperature. This mission will demonstrate high spatial resolution mapping of sea surface temperature and application to circulation studies and modeling, fog prediction, upper ocean forecasting, and fisheries operations. A scanning microwave radiomete complemented by a microwave scatterometer are required for this experiment.

Ocean Currents. The objective of this experiment is to develop signatures for the detection and mapping of ocean currents, eddies, and internal waves; and to further the understanding of the interaction of currents with waves. The ultimate objective is to measure magnitudes and directions of current flows. Primary sensors for this experiment are an altimeter and a microwave scatterometer.

Geoid Measurement. The intent of this mission is to map the ocean geoid in a low inclination orbit. Accomplishment will provide supplementary data and calibration for high inclination GEOS-3 and Seasat-A altimeters. A pulse compression radar altimeter is required for this mission.

Sea Ice Survey. In this mission periodics surveys of floating ice fields in ocean shipping lanes will be accomplished to determine location and extent of hazardous conditions. Passive and active imaging microwave systems accompanied by an altimeter and a camera constitute the desired instrument complement for this mission.

Storm Assessment. This experiment is directed at measuring the strength of tropical storms - surface winds, liquid water and water vapor, surface temperature, and wave fields - to determine landfall damage/erosion and storm surges. Active and passive microwave imaging systems and an altimeter are required for this experiment. A camera is desirable as an additional sensor if available.

Cloud Climatology. The intent of this mission is to gather baseline data of cloud properties to a geographic scale of 200 km and a temporal scale covering both diurnal and seasonal variations. The observing system consists of both an active and a passive instrument: the laser ranging system and the cloud physics radiometer.

Ozone Mapping. This mission will provide supplementary baseline data for determing ozone depletion, and serve as a calibration source for operational ozone monitoring sensors on free flying satellites. A backscatter ultraviolet spectrometer is required for this mission.

Adaptive Multibeam Communications. The AMPA experiment is specifically planned to demonstrate the feasibility of low power, point-to-point communication at L-band via low orbiting spacecraft using adaptively formed narrow beams. Multiple modes of operation-ship to ship, ship to shore, and land mobile platforms - are planned to demonstrate feasibility. A phased array antenna system is the essential Spacelab hardware required for this experiment.

2.2.2 SYNERGISTIC PAYLOAD BENEFITS

From a snyergistic standpoint, the experiments included within this payload provide many opportunities for enhanced information. This synergism occurs for both intradiscipline experiments as well as cross-discipline combinations. Examples of payload synergism are provided in the following paragraphs.

The Urban and Regional Planning, Range Assessment, and Timber Inventory missions are all land area delineating processes. Each may contribute data to regional land use inventories or may interact with regard to establishing boundaries.

One of the key facets of the Water Inventory mission is the determination of soil moisture. Grouping the Water Inventory, Crop Survey, Range Assessment, and Vegetation Stress missions on the same payload provides an opportunity for determining the effect of soil moisture on vegetation stress, range condition and crop prediction. In addition, the Crop Survey may locate areas of vegetation stress for study in that program.

It is apparent that Sea Surface Temperature and Ocean Currents have effects on marine life habitat. Combining these missions will increase the knowledge of this interaction and lead to improved methods for management of fishery and crustacean food sources.

Flying the Cloud Climatology mission provides cloud cover information which can be used for correction of microwave instrument data. In addition, it may be possible to use the cloud cover information adaptively in real time to change target areas for those missions/ experiments employing optical sensors operating in the visible portion of the spectrum.

2.2.3 EQUIPMENT COMMONALITY

Commonality of equipment for the EVAL experiments included within this payload is shown in Table 2-2. From this chart it can be seen that almost all of the sensors have application in more than one experiment, and within more than one discipline. In particular, instruments such as the thematic mapper, large format camera, and active or passive microwave imagers are required, or desired, for over half of the experiments.

Table 2-2. Full Spacelab EVAL Mission/Sensor Matrix

Sensors \ Missions	Crustal Motions	Sea Surface Temperature	Ocean Currents	Geoid Measurement	Sea Ice Survey	Storm Assessment	Crop Survey	Vegetation Stress	Urban Planning	Timber Inventory	Range Inventory	Mineral Exploration	Marine Resources	Water Inventory	Cloud Climatology	Multibeam Communications	Ozone Mapping
Shuttle Imaging Microwave System (SIMS)		(X)	X		(X)	(X)	(X)	(X)			X		X	X			
Shuttle Imaging Radar (SIR/SAR)			(X)		(X)	(X)	(X)	(X)		(X)	(X)	(X)	(X)	(X)			
Multi-Spectral Scanner (Thematic Mapper)	X	X	(X)				(X)	(X)	(X)	(X)	(X)	(X)	(X)	(X)			
Large Format Camera (LFC)	X				X	X	(X)	(X)	(X)	(X)		(X)					
GEOS-C Altimeter (1m dish)			(X)	(X)	X	(X)											
Spaceborne Laser Ranging System (LRS)	(X)														(X)		
Cloud Physics Radiometer (CPR)															(X)		
Solar Backscatter UV Spectrometer/Total Ozone Mapper (SBUV/TOMS)																(X)	
Adaptive Multibeam Phased Array (AMPA)															(X)		

(X) Required

X Desired

2.2.4 EVAL PAYLOAD

A layout drawing of the complete payload integrated for this flight is provided in Figure 2-13. The Shuttle Imaging Radar and the Adaptive Multibeam Phased Array are colocated on the aft pallet, and deployed out opposite sides of the Orbiter for operations throughout the on-orbit portion of the flight. The space between the aft pallet and the Spacelab pressurized module is occupied by the Shuttle Imaging Microwave System; while the remaining sensors are mounted on the STR bridge over the access tunnel between the Orbiter and Spacelab. Scientific descriptions of the EVAL sensors included in this payload are provided in Tables 2-3 and 2-4.

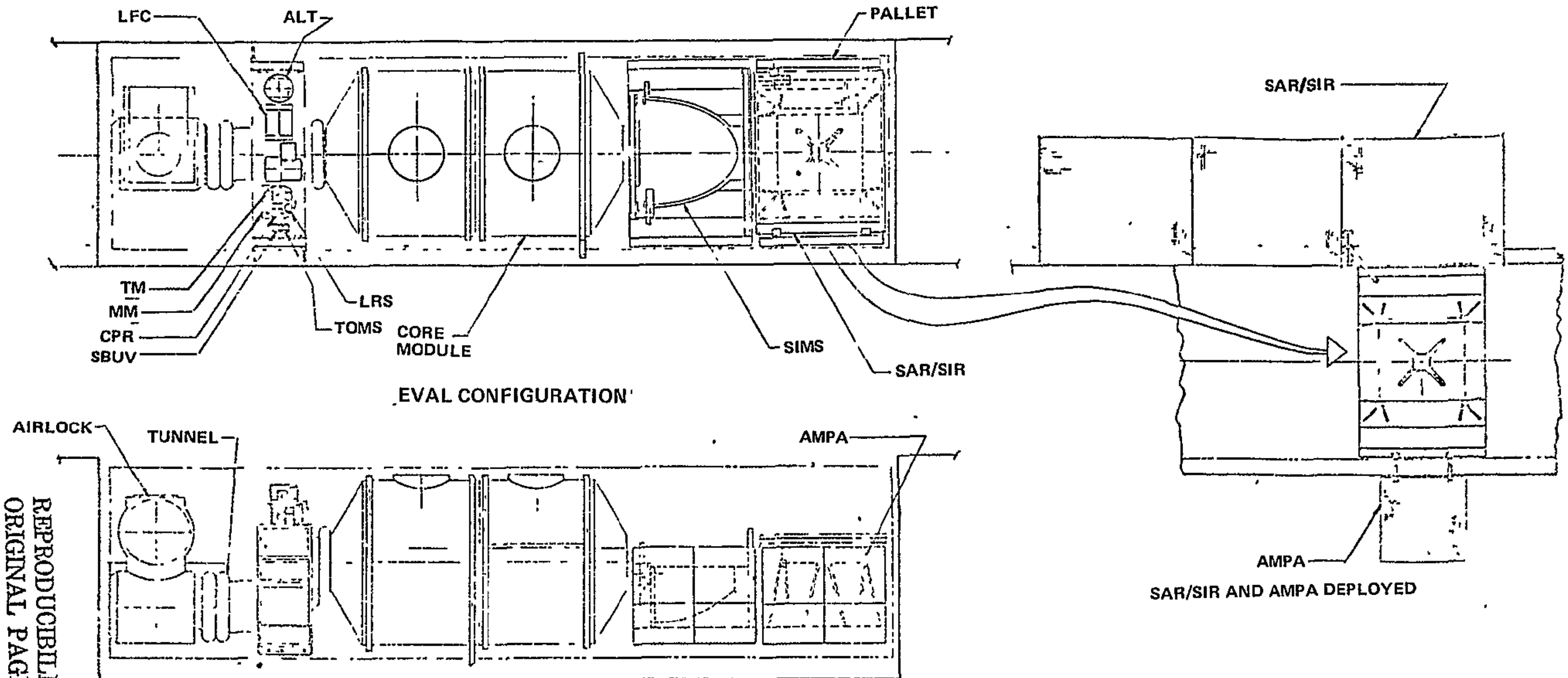


Figure 2-13. Payload Layout

Table 2-3. Sensor Characteristics

	Weight (Kg)	Stability Amplitude (sec)	Pointing Accuracy (deg)	Size (cm) L	W	H	Power (Watts) Avg	Peak	Data Rate (BPS)
Shuttle Imaging Microwave System (SIMS)	952	36	0.05	400	300	250	930	TBD	3×10^6
Shuttle Imaging Radar (SIR)	1248	6	0.1	10.7m	3m	15.5	6kW 2kW	6kW Operation 2kW Warmup for 1/2 hr.	480×10^6
Thematic Mapper (TM)	180	6	0.5°	116	93	60	55	80	120×10^6
Large Format Camera	136	3	0.5	79	64	73	120	500 for 10ms per frame	N/A
Altimeter GEOS-C Electronics	68	72	0.1	100 56	100 64	15 64	150	150	15K
Cloud Physics Radiometer (CPR)	187	—	0.2	81	25	36	25	25	500K
Laser Ranging System (LRS)	60 200	0.2 mrad	2 mrad (knowledge)	82 $1M^3$	57 Electronics	36	250	—	50K
SBUV/TOMS Electronics	20	TBD	0.5°	53 33	38 15	21 20	15	19	320
AMPA Antenna	348	0.1 deg/sec	0.5°	Pallet 350	280	250	750	800	1×10^6
Electronics	75			Rack 48	36	152		500 standby	

Table 2-4. Sensor Capabilities

	Type	Objective	Spectral Bands		Viewing	
			Number	Range	Direction	Field of View
Shuttle Imaging Microwave System (SIMS)	Passive Microwave Radiometer	Measure Thermal Emission from Earth's Surface	11	0.61 - 118.7 GHz	Nadir	60° Cross Track 17° Along Track 0.9°-17° Instantaneous
Shuttle Imaging Radar (SIR)	Active Microwave	Obtain High Resolution Microwave Imaging	2	9.0 GHz 1.04 GHz	Sidelooking (10° from Nadir to 60°)	21°
Thematic Mapper (TM)	Scanning Spectral Radiometer	Obtain High Resolution Multispectral Imaging	4 1 1 1	0.5 - 1.1 μm 1.55 - 1.75 μm 2.1 - 2.35 μm 10.1 - 12.6 μm	Nadir $\pm 20^\circ$	14° A_Z, 2° El Total, $.0017^\circ$ A_Z, $.0068^\circ$ El Instantaneous
Large Format Camera	Framing Mapping Camera	Provide High Resolution Stereo Photography	1	0.5 - 0.85 μm	Nadir	40° Cross Track 80° Along Track
Altimeter GEOS-C	Active Microwave	Measure Altitude above Ocean and Terrain	1	13.9 GHz	Nadir	1.5°
Cloud Physics Radiometer (CPR)	Scanning Imaging Radiometer	Measure Cloud Temperature and Water Content	8	0.75 - 10.99 μm	Nadir	90° Total 0.4° Instantaneous
Laser Ranging System (LRS)	Active Optical	Measure Range from known Position to Unknown Position	1	Nd: Yag	Nadir	130° Total 0.028° Instantaneous
SBUV/TOMS	Spectral Radiometer	Measure Solar Irradance	12	160 - 400 μm	Nadir	11.3°
AMPA Antenna	Active/Passive Microwave System	Demonstrate Low Power Point-to-Point Communications	3	1.5, 1.6 GHz Active 1.4 GHz Passive	Nadir $\pm 30^\circ$	$\pm 40^\circ$ about Viewing Direction

2.3 PHYSICAL ACCOMMODATIONS

The physical accommodation of payload equipment on the Spacelab pallet, in the pressurized module, on the STR bridge, and in special installations presents a multi-faceted challenge to the payload designer. Available volumes and areas are limited, field of view requirements are often conflicting, and weight and volume constraints can be critical. The dedicated EVAL payload was selected with the intent of exercising Shuttle/Spacelab capabilities to the fullest, and in so doing uncover problem areas in accommodations and operation. As a result, valuable insight is gained into the realities of Shuttle/Spacelab utilization for earth viewing missions.

2.3.1 PAYLOAD WEIGHTS AND LOCATIONS

Payloads and payload chargeable weights of experiments, experiment support equipment, carriers (Spacelab and STR), excess crew, mission extension kits, and contingency allowance are summarized in Table 2-5. The payload launch weight of 15,317 kg noted in this table is well below the allowable launch weight of ~ 25,000 kg associated with the launch conditions specified for this flight. The landed weight of 14,451 kg is only 64 kg below the 14,515 kg landing weight limit.

Spacelab and STR weights are broken down in Table 2-6. Mission dependent subsystems (consisting of Spacelab racks and other mounting structure; habitability equipment; EPDS, C&DMS, and ECS equipment; common payload support equipment; and a Spacelab weight reserve) are estimated to weigh 1379 kg based on the weight budget for Spacelab 1. Weights for mission independent subsystems, the transfer tunnel, and mission independent Orbiter support are the latest available Spacelab element mass properties values (8/10/76).

Significant features of this payload are:

1. The CPR and LRS are installed on a small stabilized platform, Minimount, which is attached to the port side of the STR bridge. For a morning launch and a nose forward, inverted (X-IOP, Z-LV) attitude, the port side is the sunlit side.

2. The SIMS structure is mounted between the Spacelab module and Spacelab pallet in order to move payload c.g. as far aft as possible.

Table 2-5. Payload and Payload Chargeable Weights

	Launch Weight (kg)	Landed Weight (kg)
Experiment Sensors	3199	3199
TM	(180)	(180)
LFC	(136)	(136)
ALT	(68)	(68)
SBUV/TOMS	(20)	(20)
CPR	(187)	(187)
LRS	(60)	(60)
SIMS	(952)	(952)
SIR	(1248)	(1248)
AMPA	(348)	(348)
Experiment Support Equipment	1585	1585
Minimount	(200)	(200)
VHDRR	(230)	(230)
OEDSF	(115)	(115)
CC Electronics	(250)	(250)
SSA Electronics (SIMS, SIR, AMPA)	(600)	(600)
Misc. Expt. Support Equipment	(190)	(190)
Other Payload Changeable Weight	1263	876
Crew Eqpt and Consumables (above baseline)	(268)	(268)
Electric Energy Kits (above baseline)	(756)	(369)
Payload Weight Contingency	(239)	(239)
Spacelab and STR	9270	8791
Mission Independent Subsystems	(5723)	(5631)
Mission Dependent Subsystems	(1379)	(1379)
Transfer Tunnel	(428)	(428)
Orbiter Support Equipment	(1377)	(990)
STR	(363)	(363)
Total Payload Weight at Launch	15317	
Total Payload Weight at Landing		14451
Payload Weight Margin at Launch *	9683	
Payload Weight Margin at Landing **		64

* Based on 25,000 kg launch weight capability

** Based on 14,515 kg landing weight limit

Table 2-6. Spacelab and STR Weight

	Launch Weight (kg)	Landed Weight (kg)
Mission Independent Subsystems	5723	5631
Module	(4690)	(4598)
Pallet	(700)	(700)
Utility Harness (Forward)	(236)	(236)
Payload Specialist Station	(97)	(97)
Transfer Tunnel	428	428
Tunnel/Air Duct	(428)	(428)
Mission Dependent Subsystems	1379	1379
Racks, RAUs, EPDS Eqpt., etc.	(1379)	(1379)
Mission Independent Orbiter Support	1377	990
Electrical Energy Kit (baseline)	(756)	(369)
Heat Rejection Kit	(88)	(88)
Retention Fittings (1 set)	(125)	(125)
Tunnel Adapter	(408)	(408)
STR	363	363
Bridge Structure	(313)	(313)
Support Subsystems	(50)	(50)
Total Spacelab and STR Weights	9270	8791

3. The SIR and AMPA deploy over opposite sides of the Spacelab pallet. The SIR then unfolds forward to its full open position for experiment operations. It can be refolded when not operating so that Orbiter radiator shielding is minimized.

2.3.2 PAYLOAD CENTER OF GRAVITY

The aerodynamic flight phases of the Shuttle Orbiter (entry and landing, boost phase abort) place rigid center of gravity constraints on Shuttle payloads. The most severe are the X-axis limits which require payload c.g. to be in the aft portion of the payload bay, and the Y-axis limits which require payload c.g. to be within a few inches of the payload bay centerline. Z-axis limits are less stringent, allowing c.g. locations up to 4 feet above or below the payload bay centerline.

All payload chargeable items are included in c.g. determination, including payload equipment in the Orbiter Aft Flight Deck and crew consumables, crew equipment, and mission extension kits over and above baseline allowances. For a Spacelab mission, all mission independent and mission dependent equipment is payload chargeable, including the Transfer Tunnel and Tunnel Adapter. The Orbiter Airlock is not payload chargeable.

Payload c.g. locations for the dedicated EVAL mission are shown in Table 2-7. The unballasted payload c.g. falls just outside the X-axis limit for both landing and launch (see Figure 2-14). Adding 1000 kg of ballast on the aft pallet moves longitudinal c.g. within its acceptable envelope, but landed payload weight exceeds the allowable landing limit by 936 kg. Thus, a choice must be made between violating one or the other of the design constraints, or degrading the mission by eliminating part of the payload. Because the c.g. envelope is a conservative estimate that allows for a range of Orbiter c.g. locations (the significant parameter is combined payload and Orbiter c.g.), and the landing weight limit is a design point and not a true limit at all; for this study the dedicated EVAL mission has been left as originally defined on the assumption that c.g. can be made acceptable with a small amount of ballast. However, early planning for real missions should probably incorporate the groundrule that payloads must fall well within given system constraints, with reasonable margins left over to accommodate unforeseen growth. Therefore, the dedicated EVAL mission has been left as originally defined, and the assumption is made that c.g. can be made acceptable with a small amount of ballast.

Table 2-7. Payload Center of Gravity

	Weight (kg)	X_{cg} (m)	Y_{cg} (m)	Z_{cg} (m)
<u>Experiment Sensors</u>				
TM	180	3.80	-0.35	1.20
LFC	136	3.80	0.75	1.40
ALT	68	3.80	1.60	1.15
SBUV/TOMS	20	3.80	-1.75	1.10
CPR	187	3.75	-1.10	1.60
LRS	60	4.00	-1.10	1.50
SIMS	952	13.50	0	-1.15
SIR	1248	16.65	0	0.55
AMPA	348	16.65	0	0.18
<u>Experiment Support Eqpt.</u>				
Minimount	200	3.80	-1.10	1.10
VHDRR	230	8.75	-1.27	0.18
OEDSF	115	8.75	-1.27	0.18
CC Electronics	250	8.75	1.27	0.18
SSA Electronics (SIMS, SIR, AMPA)	600	10.00	0	0.18
Misc. Expt. Support Eqpt.	190	12.00	0	0
<u>Other P/L Weights</u>				
Crew Eqpt	268	-0.81	0	1.50
Energy Kit (Launch)	756	11.40	0.14	-2.57
Energy Kit (Landing)	369	12.29	-0.52	-2.59
P/L Contingency	239	12.00	0	0
<u>Spacelab and STR</u>				
Miss. Ind. Syst. (Launch)	5723	9.20	0.02	-1.50
Miss. Ind. Syst. (Landing)	5631	9.20	0.02	-1.50
Miss. Dep. Syst.	1379	9.75	0	0
Transfer Tunnel	428	3.80	0	-0.50
Orbiter Support Eqpt (Launch)	1377	7.26	0	0
Orbiter Support Eqpt (Landing)	990	7.26	0	0
STR	363	3.80	0	-0.70
Center of Gravity at Launch	15317	9.51	-0.018	-0.225
Center of Gravity at Landing	14451	9.45	-0.040	-0.091
Ballast	1000	17.50	0	-1.60
Center of Gravity at Launch	16317	10.00	-0.018	-0.310
Center of Gravity at Landing	15451	10.01	-0.040	-0.189

Payload c.g. locations in the Y and Z axis directions are well within limits for both the ballast and unballasted case (Figures 2-15 and 2-16).

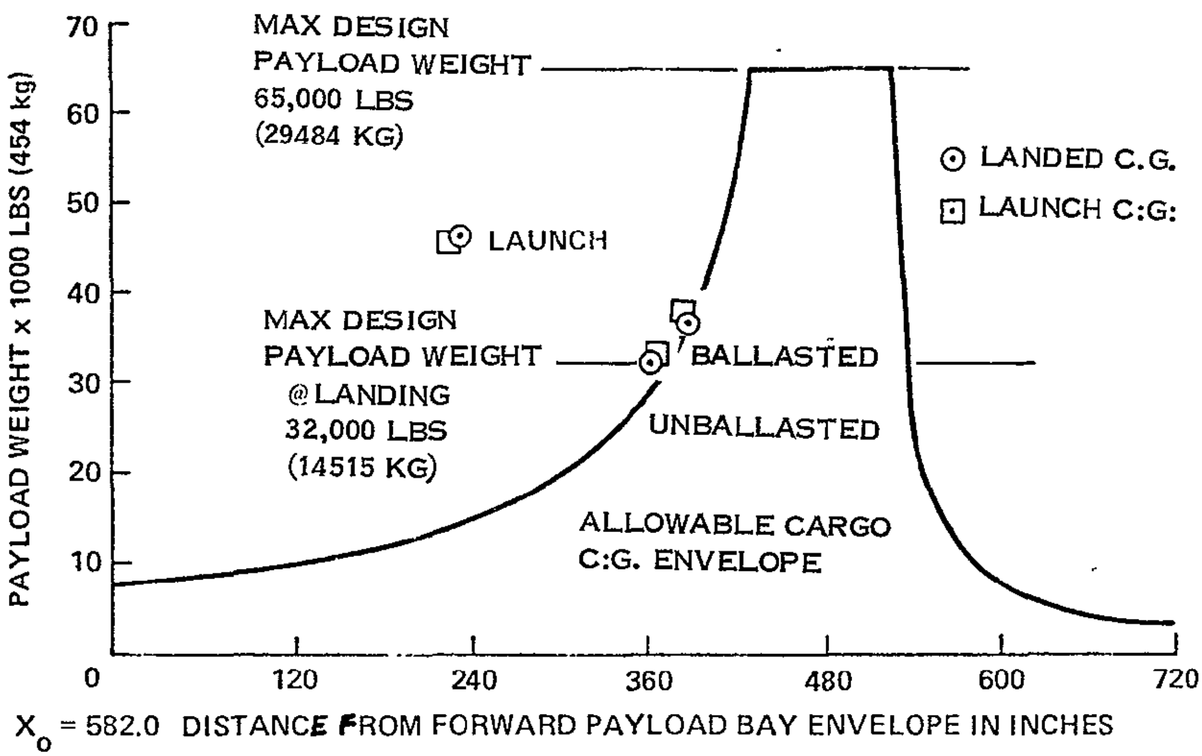


Figure 2-14. Payload CG Along X-Axis

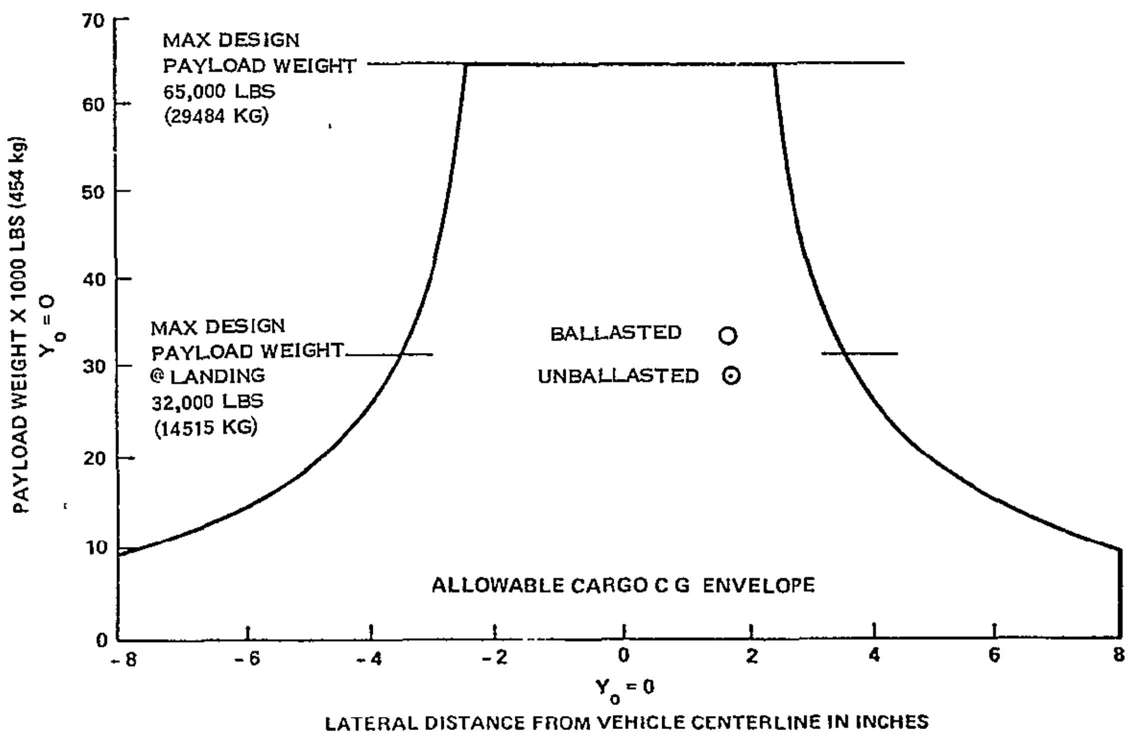


Figure 2-15. Payload CG Along Y-Axis

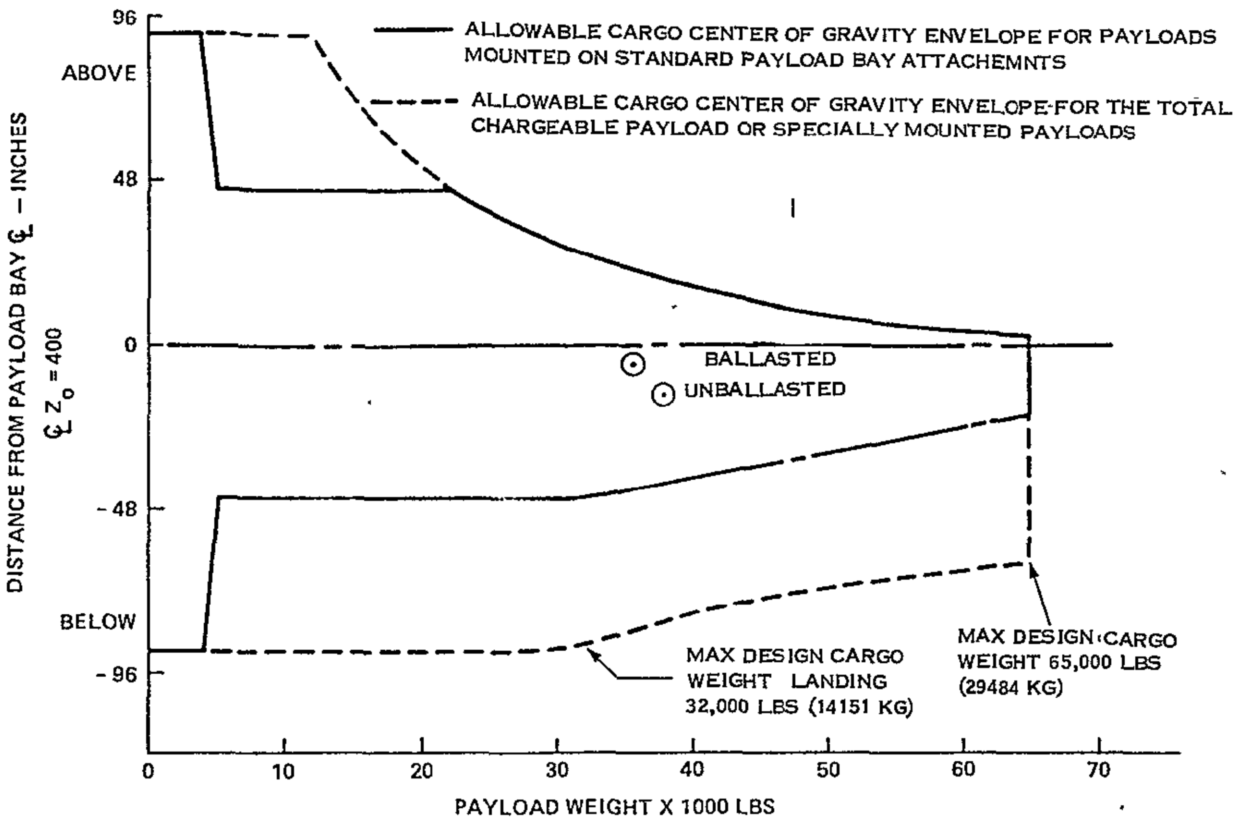


Figure 2-16. Payload CG Along Z-Axis

2.3.3 PRESSURIZED VOLUME

The long module (Core Segment plus Experiment Module) configuration provides 14.10 m^3 of payload volume in six double racks and two single racks. Table 2-8 indicates that only about 50% of this capability is required for the dedicated EVAL payload. This is at best an estimate - the Control and Display (C&D) and electronic support requirements of most experiments are not defined at present. It would appear, however, that ample pressurized volume is available for this payload.

The total weight capability of the long module racks is 4640 kg. The currently identified weight of pressurized equipment is 1258 kg, which is well within this limit. Hence, ample payload weight capability is available for pressurized equipment, except as constrained by total weight margin and c.g. requirements.

Table 2-8. Pressurized Equipment

Equipment	Weight	Volume	Remarks
VHDRR	230 kg	.42 m^3	Best available information
OEDSF	115 kg	.17 m^3	Conservative for 2-array system
CC Electronics	250 kg	1.80 m^3	Estimated
SSA Electronics	600 kg	4.30 m^3	Estimated
Other Electronics	63 kg	.45 m^3	1/3 of Misc. Expt Support Eqpt.
Total Pressurized	1258 kg	7.14 m^3	

2.3.4 FIELD OF VIEW

An assessment of EVAL experiment fields-of-view (FOV) appears in Table 2-9. The EVAL payload arrangement satisfies all experiment viewing requirements with the following provisions:

1. The Cloud Climatology sensors lose one quadrant of their viewing cover due to obstruction by the TM.
2. The AMPA antenna loses a small portion of its viewing cover due to obstruction by the Orbiter vertical tail.

The SBUV/TOMS can look at the sun for calibration at each and every orbital dawn and dusk throughout the mission. The time during which the sun is visible (above the horizon and below the open cargo bay doors) varies from about 7 minutes early in the mission to less than 5 minutes late in the mission. This variation is due to changing β angle. The sun is always seen off the port side of the orbiter, in the forward quarter at dawn and the aft quarter at dusk. Viewing azimuths (measured aft from the orbiter X-axis) vary from about 45^o at dawn and 135^o at dusk early in the mission to 30^o at dawn and 150^o at dusk late in the mission.

Table 2-9. Experiment Field of View Assessment

Experiment	Viewing Requirement	Viewing Capability
TM	Nadir Viewing $\pm 20^o$ Offset (Cross Track) $0.0017^o \times 0.0068^o$ Inst. FOV 14^o Total FOV (Cross Track) 2^o Total FOV (Along Track)	Location in center of STR provides unobstructed nadir view and 20^o offset pointing to either side
LFC	Nadir Viewing $40^o \times 80^o$ Inst. FOV 40^o Total FOV (Cross Track) 80^o Total FOV (Along Track)	Location near center of STR provides unobstructed nadir view
ALT	Nadir Viewing 1.5^o Inst. FOV 1.5^o Total FOV	Location on stardboard side of STR provides unobstructed nadir view
SBUV/TOMS	Nadir Viewing Solar View (Full Sun) for Cal 11.3^o Inst. FOV 11.3^o Total FOV	Location on port side of STR provides unobstructed nadir view and full sun viewing for 5 to 6 minutes at dawn and dusk
CPR	Discrete Targets (Cloud Tops) $\pm 65^o$ off Nadir (conical) 0.4^o Inst. FOV 0.4^o Total FOV	Minimount near center of STR provides up to 65^o offset pointing fore and aft and to the port side. Pointing to the starboard side is limited to less than 20^o by TM
LRS	Discrete Targets (Cloud Tops) $\pm 65^o$ Off Nadir (Conical) 0.03^o Inst. FOV 0.03^o Total FOV	Minimount near center of STR provides up to 65^o offset pointing fore and aft and to the port side. Pointing to the starboard side is limited to less than 20^o by TM
SIMS	Nadir Viewing 0.9^o to 17^o Inst. FOV (Fuction of Frequency) 60^o Total FOV (Cross Track) 17^o Total FOV (Along Track)	Location in first pallet position provides unobstructed nadir view
SIR	Side Viewing 7^o to 60^o Off Nadir 21^o Inst. FOV 21^o Total FOV	Location on starboard side of second pallet provides unobstructed view to starboard
AMPA	Nadir Viewing $\pm 30^o$ Off Nadir (Conical) $\pm 40^o$ Inst. FOV $\pm 40^o$ Total FOV	Location on port side of second pallet provides up to 70^o viewing in all directions except where obstructed by the Orbiter's vertical tail

2.3.5 INTERFACES

Payload to Shuttle/Spacelab interface definition is begun with schematic diagrams that define the payload accommodation resources utilized by each experiment. Two examples of these experiment schematics are given in Figures 2-17 and 2-18. The first figure shows required connections between the CPR sensor and the STR mounting system; while the second shows connections between the SIR experiment and the Spacelab module and pallet. Electric power, command/telemetry, data, C&W, thermal control, mounting, and pointing system connections are defined.

The experiment schematics identify the experiment to Shuttle/Spacelab interfaces that must be designed. For example, the pallet mounted SIR equipment must tie into pallet hard points because of its large size and weight. (Smaller equipment can mount directly to pallet floor or skin panels.) Spacelab unregulated dc power can be used if the experiment design incorporates the proper power conditioning/supply equipment. Provisions must be made to route experiment data through the Spacelab high rate digital channels. Caution and warning circuits are required to monitor antenna deployment and retraction.

On the STR, the CPR equipment requires a pointing system such as Minimount. Electric power will be used as provided, and all experiment data will be recorded. High voltages in the CPR electronics may require C&W monitoring. Active thermal control (ATC) may be required; and if so, it can be provided by the Minimount canister.

Once the individual experiment interfaces have been identified, the combined payload to Shuttle/Spacelab interfaces can be investigated. This is accomplished with a system schematic as shown in Figure 2-19. Payload equipment is assigned a specific rack, pallet, or other location; and all required connections are shown. Each connection is analyzed to ensure that combined payload requirements are compatible with the payload accommodation capabilities of each carrier element (rack, pallet, STR, etc.). When compatibility is ensured, detailed interface design can proceed.

The dedicated EVAL payload shows no interface incompatibilities. The thematic mapper requires a special line to transmit very high rate data to the VHDRR in the Spacelab module.

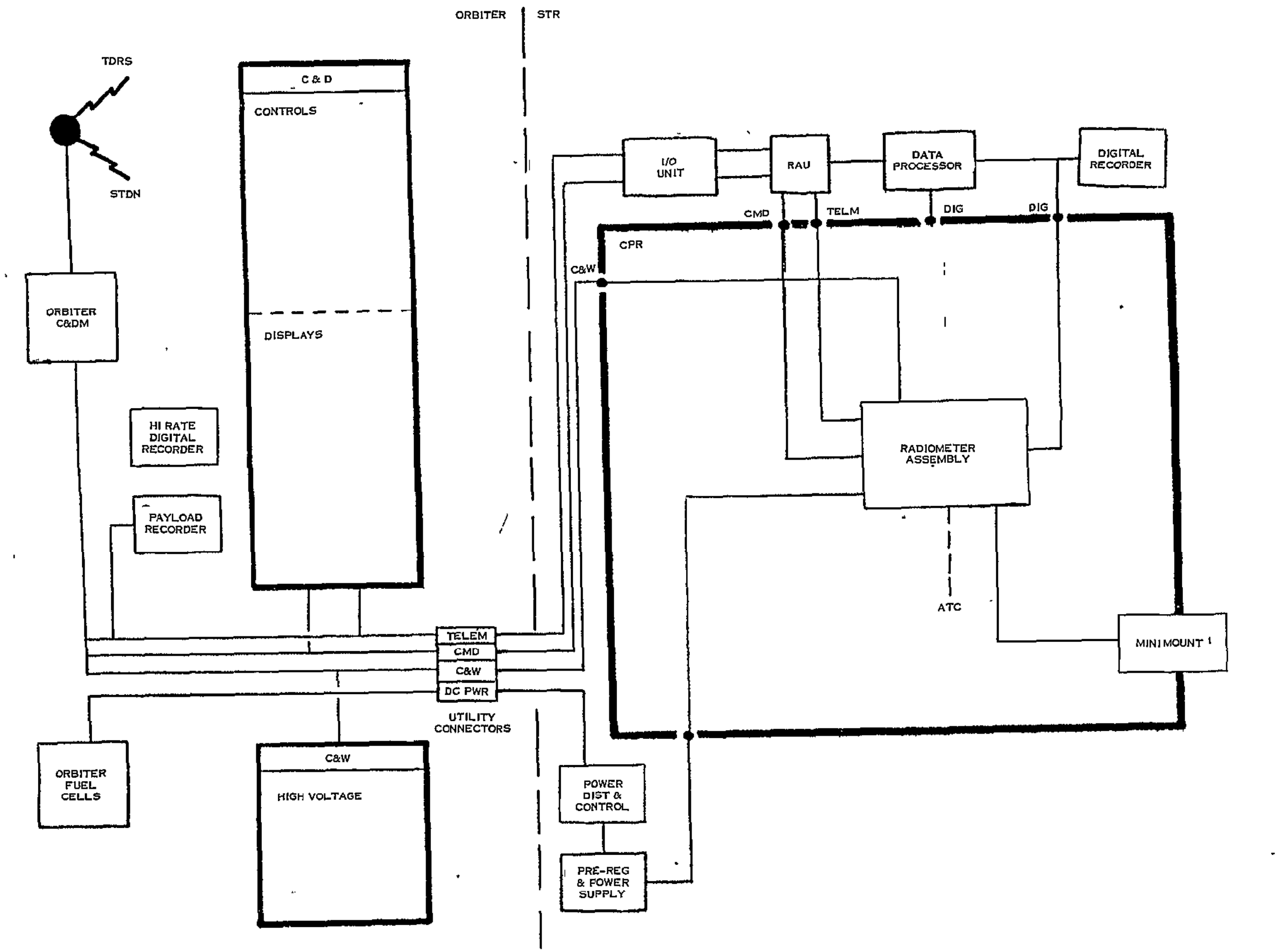


Figure 2-17. EVAL Experiment Schematic - CPR

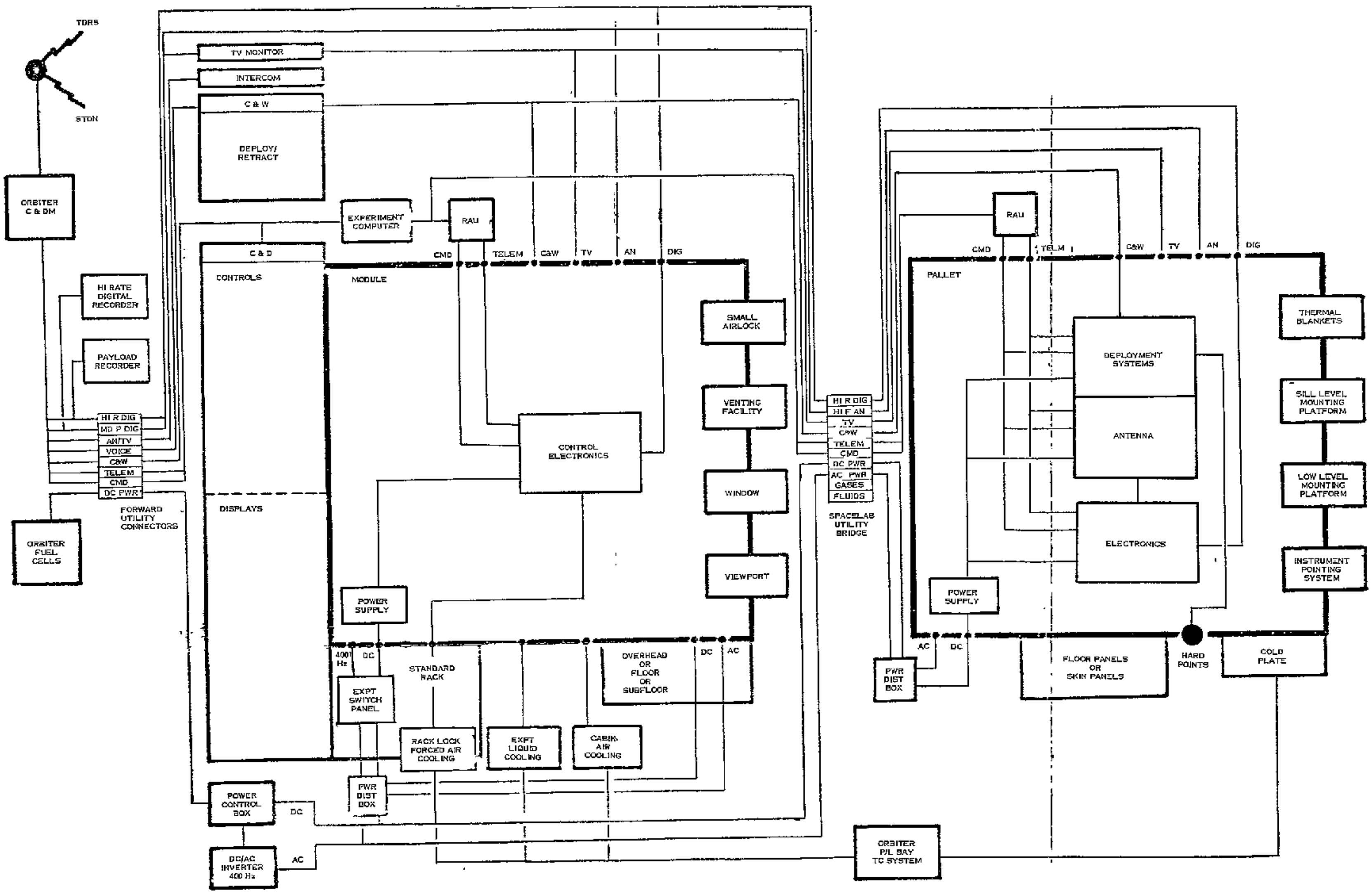


Figure 2-18. EVAL Experiment Schematic - SIR

This line uses available capability in the forward end cone feedthrough panel. Electric power, command/telemetry, and caution and warning connections between STR and Orbiter are routed through utility service panels on the forward bulkhead of the cargo bay (Sta 576) and on the starboard sidewall (Sta 695). These locations are shared with Spacelab, resulting in a common power bus and a common data (Command/telemetry) bus for STR and Spacelab. The SIMS experiment is located between the Spacelab module and pallet, so utilities must be routed through or around it. This is accomplished by using standard Spacelab cabling and utility bridges to make what is essentially no more that an extra long module-to-pallet connection.

2.4 OPERATIONS

2.4.1 EXPERIMENT OBSERVATIONS

An assessment of on-orbit mission operations related to the dedicated EVAL payload has been performed to determine experiment observation periods, crew requirements and timelines, and profiles for mission resources such as power and data.

The approach involved fitting the requirements of the various experiments/missions to the orbital conditions of the flight in the most judicious manner. Initially, earth oriented target locations, both point and area, were identified for the specific experiments and spotted on a global map. Table 2-10 correlates this data along with lighting and operation requirements for each experiment.

Next, orbits were run for a sortie mission having the specified conditions of 200 km altitude and 57° inclination; and assuming a launch from the ETR at Cape Kennedy. An on-orbit flight duration of 7 days (155 hours) was planned based on the payload weight analysis described in the preceding section. Orbit eccentricity and decay rate are both specified as zero, and injection is assumed to be over Cape Kennedy at the time of launch for simplicity. The launch time and data were selected at 0700 Eastern Standard Time on the 15th of August, 1982 (the prescribed month and year) to ensure significant daylight observation time over CONUS, the North Atlantic, and the North Pacific - which are prime target areas for many of the experiments. As a consequence, the southern hemisphere is generally overflown at night.

Recovery was accomplished on Orbit 119 on a Northwest to Southwest pass which essentially flies directly over the Cape Kennedy recovery area. Estimated landing time is approximately 13:25 Eastern Standard Time.

From the orbit calculations, ground tracks are obtained which indicate which orbits overfly the various target areas. A sample of these ground tracks for a typical one day time frame, approximately 16 orbits, is shown in Figure 2-20.

Table 2-10.. Earth Resources Test Sites

Experiment	Target	Lighting (Sun Angle)	Operation
Crustal Motions/Land Subsidence	Southern Flordia, California	Day	As many passes as possible - from 20^{o} elevation angle thru nadir to 20^{o} elevation
Ocean Currents	Sea of Japan	Day	2 passes minimum in each area
Sea Surface Temperature	Grand Banks, Spanish Sahara Coast, Peruvian Coast	Day	2 passes minimum in each area
Geoid Measurement	Global	Day or Night	Continuous for 3 revolutions
Sea Ice Survey	Global Ocean Area at Latitudes > 55^{o} (Cape Horn)	Day or Night - Day preferably	Once per day in major shipping areas
Storm Assessment	Global Ocean Areas ($\pm 25^{o}$ Latitude) Except South Atlantic	Day or Night - Day preferably	As many passes as possible
Crop Survey	See Table 2- a.	Day (10:00 - 14:00)	2-3 passes over each target
Vegetation Stress	CONUS - See Table 2- a	Day (10:00 - 14:00)	2-3 passes over each target
Urban Planning	CONUS - See Table 2- a	Day	2-3 passes over each city
Timber Inventory	CONUS - See Table 2- a.	Day	2-3 passes over each area
Rangeland Status	CONUS - See Table 2- a	Day (10.00 - 14.00)	2-3 passes over each target
Mineral Exploration	S.W. CONUS,	Day (08·00 - 17·00)	2-3 passes over each area
Marine Resources	See Table 2- a	Day (10:00 - 14:00)	2-3 passes over each target
Water Inventory	See Table 2- a	Day (10·00 - 14 00)	2-3 passes over each target
Cloud Climatology	Global (0 to $\pm 15^{o}$ Latitude, 30 to 50^{o} N&S Latitude	Day or Night	As many passes as possible - 10 minutes duration on each pass
Multibeam Communications	CONUS and Broad Ocean Areas, STDN Stations	Day or Night	2 to 8 operations plus 2 calibrations (STDN Stations) per day - 3 minutes duration for each operation
Ozone Mapping	Global (20 to 50^{o} N&S Latitudes Over Continental Areas)	Day	4 observations of 15 minutes duration each, plus 2 sun calibrations

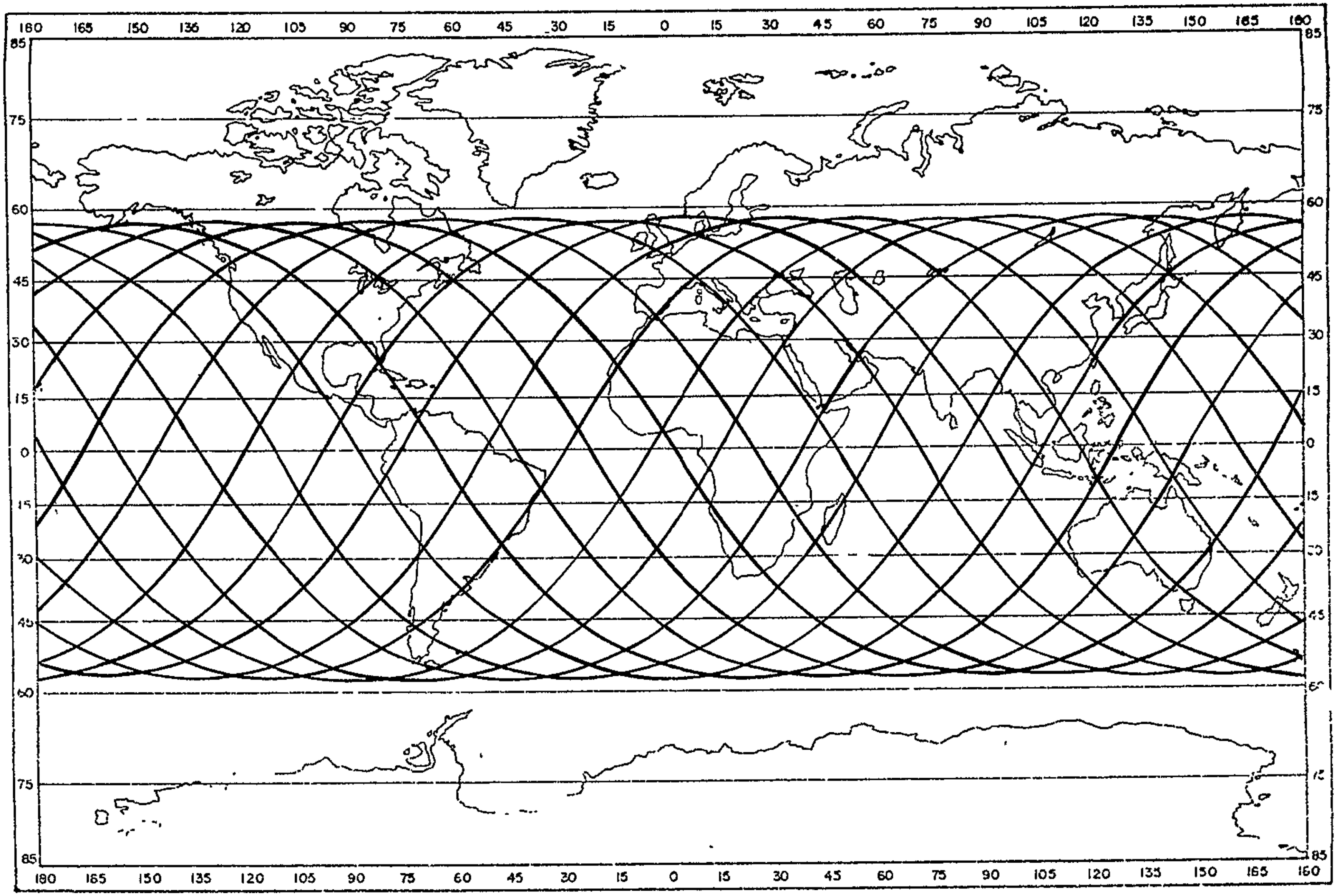


Figure 2-20. Typical One Day (Orbits 30 to 46) EVAL Mission Ground Trace

Using the orbit ground tracks as input data, specific target areas for each mission were chosen on an iterative basis. That is, the group of candidate test sites was modified several times to accommodate the orbit and the sensor package. The viewing angles of the Thematic Mapper and the SIR were important limitations, since they frequently are required to view the same target on the same pass. (Their characteristics and locations result in an overlapping field of view of only 17^{o}, situated between 10^{o} and 27^{o} from nadir on the port side of Shuttle's ground track.) The final list of test sites presented is representative of the complete list of target areas of interest, and offers an excellent opportunity for successfull applications development.

The specific test sites selected for each mission are identified in the following paragraphs, and the rationale for their selection explained. In many cases, the test site selection was as much a function of the spacecraft orbit as any other selection criterion.

Timber Inventory Mission

Five national forest areas were selected for test sites for the Timber Inventory Mission. These sites are currently well surveyed and represent a comprehensive and representative sample of the various forest types found in theU.S. (See Table 2-10a.) The summer schedule and fairly high sun angle of the specified orbit over these test areas are conductive to accurate timber classification. The primary output of the timber inventory mission will be tabular data summaries of estimated harvestable timber volume.

The acquisition and classification of orbital data will be the first stage of a multi-stage probabilistic sampling strategy. The first stage is intended to classify timberlands according to volume strata and to select sampling sites for the next stage. Subsequent stages will employ aircraft overflights and ground crew measurements to establish volumetric measurements for each stratum.

Urban and Regional Planning Mission

Five cities were selected as representative test sites for the Urban and Regional Planning Mission. They are Miami, Florida, New Haven, Conn., Birmingham, Ala., El Paso, Texas,

Table 2-10a. Target Requirements

Category	Area	Longitude	Latitude	Remarks
1. Timber Inventory	Williamette Nat For.	121.5-122.5	43.5-44.5	
2. Timber Inventory	Apache Nat. For.	108.3-109.4	33-34	
3. Timber Inventory	Green Mtn. Nat. For.	72.7-73 3	42.7-44	
4. Timber Inventory	Talladega Nat. For.	85.7-86.3	33-33.7	
5. Timber Inventory	Clark Nat. For.	90.8-91.4	37.3-38	
6. Land Use Inventory	Miami, Fla.	80.1	25.8	
7. Land Use Inventory	New Haven, Conn.	72.9	41.3	
8. Land Use Inventory	Portland, Oregon	122.6	45.5	
9. Land Use Inventory	El Paso, Texas	106.0	31.75	
10. Land Use Inventory	Birmingham, Ala.	86.8	33.5	
11. Minerals Exploration	Arizona	110-111.5	35 5-37	
12. Minerals Exploration	Nevada	114-115.5	38.5-40	
13. Minerals Exploration	U.S.S.R.	67°E	57°N	
14. Crop Stress Analysis	Iowa	93.5-95	40.7-42.5	
15. Marine Resources	Oregon	124-126	46-47	
16. Marine Resources	Somali	47°E	3°N	
17. Marine Resources	Peru	Entire Coastline		
18. Range Inventory	Florida	80.8-81.8	27.2-28	
19. Range Inventory	Texas	104-105	31-32	
20. Range Inventory	Nevada	116-117	41-42	
21. Water Availability	Florida	80.5-81.3	25-26.5	
22. Water Availability	Tennessee	87.5-88.5	34.3-36.3	
23. Water Availability	Arizona	(109.5-34) (111-33)	(112-36) (113-34.5)	
24. Water Availability	Oregon	119.5-121	43.6-44.6	
25. World Crop Survey	Alabama/Georgia	84.5-85.5	31-32	
26. World Crop Survey	Missouri	89-90.5	36-37.6	
27. World Crop Survey	Iowa	95.7-96.7	42.3-42.9	Plymouth & Woodbury Counties
28. World Crop Survey	South Dakota	96.5-96.8	42.5-43.1	Clay & Union Counties
29. World Crop Survey	South Dakota	96.3-97.5	44-45	
30. World Crop Survey	People's Republic of China	119-120E	30-31N	
31. World Crop Survey	U.S.S.R.	50-51E	48-49N	
32. World Crop Survey	Spain	1-2W	38-39N	
33. World Crop Survey	India	74E	30N	

and Portland, Oregon. These cities represent a broad range of urban characteristics including:

Size:	population, area
Region:	physiographic, demographic
Age:	settlement, development
Site:	coastal /inland, river basin, watershed
Hinterland:	vegetative, forestation, arid zone
Socio-economy:	industry, commerce
Uniqueness:	critical areas, special features, ethnicity, etc

The high sun angle offered by the selected orbit is critical. The data processing and analysis of the acquired imagery will involve both manual and machine-aided interpretation of photographic and electronic image products. The output products will be land use thematic maps, coordinated with the needs of the various land-use planning agencies involved.

Mineral Survey Mission

In order to limit the mission to a manageable size, the objective is specified as "detection of geologic evidence of commercial grades and quantities of copper bearing ores." Test sites to be evaluated include known copper-producing regions and adjacent areas in order to utilize existing ground truth and proven exploratory techniques. Two areas in the United States and one foreign area (See Table 2-10a) were selected for inclusion in the program. The existence of proven copper mineralization in each area increases the likelihood that other deposits will be discovered under similar geologic conditions; thus, they are considered to be high potential exploration targets.

Variable sun angles will be available, allowing for a thorough "lineament" analysis of each area. Interpretation techniques will include both manual and computer interpretation of both the spatial and the spectral data. Results from the spatial interpretation will be in the form

of lineament maps which will indicate subsurface structure and bedding. The spectral processing, based on some of the work in the emerging field of biogeochemistry will attempt to specifically identify copper ore locations. No further processing or correlating of the data will be performed.

World Crop Survey Experiment

An extensive agricultural applications development program is currently underway, Large Area Crop Inventory Experiment Shuttle (LACIE), using data from the Landsat program. Data acquired from this Shuttle mission will be of greater resolution than Landsat data and will be accompanied by high resolution photography and Synthetic Aperture Radar data. This program will be useful for sensor parameter evaluation and for test data collection for evaluation of the various user models. Sensor parameters such as spectral band limits, over-sampling rate, SNR, etc., may be varied and the results evaluated. In addition, the added information from the higher resolution and the additional sensors will allow for investigation of the spectral and spatial classification accuracies and yield prediction user models.

Five domestic and four foreign test areas (See Table 2-10a) were selected for inclusion in this mission. A variety of crops, farming practices, field sizes, growing conditions, etc., are included in these test areas.

Vegetation Stress Detection

Due to the sporadic nature of vegetation stress, the monitoring of this phenomena over a large area is ideally suited to a Shuttle/sortie type mission. However, at this time, it is impossible to predict an area of vegetation stress due to insect infestation, drought, flooding, or disease. For this reason an area in western Iowa, which is presently being monitored by NASA's Landsat Agricultural Monitoring Program (LAMP) was selected. This area (See Table 2-10a), one of the leading corn growing regions in the world, is being closely studied for anomalous behavior. If, at the time of the Shuttle flight, a more appropriate need for this particular mission develops, schedule plans will be changed at that time. Presently, however, the LAMP test area is the primary site for this mission.

Range Condition Assessment Experiment

At some time in the future, range conditions will be continuously monitored from an orbiting satellite similarly to the world crop monitoring program. In order to achieve that goal, a substantial amount of applications development activity must be performed. This program will assist in the development of both the sensors and the user models which process the data.

Three test sites (See Table 2-10a) have been selected for this program which contain extensive range areas, and which represent three different geographic areas. These areas, made available by this orbit, offer an opportunity to advance the range management effort.

Water Availability Forecasting Experiment

Four watershed areas have been selected for study in the Water Availability Forecasting Mission (See Table 2-10a). Included among these are the Salt and Verde Watershed area in Arizona, the Tennessee River Watershed, the Crooked River Watershed in central Oregon, and the water conservation area in the Everglades of Florida. These watershed areas are being extensively studied at the present time, and will continue to be. In addition, they represent a variety of vegetation conditions, water impoundment areas, watershed sizes and land use patterns.

Living Marine Resources Assessment Program Experiment

This is the most research-oriented program of the applications missions. Its aim is to attempt to measure marine parameters such as chlorophyll, turbidity, salinity, sea state, etc., to relate these parameters to the habits and characteristics of the various species, and to develop a predictive model based on the correlation. This is an enormous undertaking and yet one that offers unlimited potential future benefits.

A current long-term study underway in this area is the Coastal Upwelling Ecosystems Analysis sponsored by the National Science Foundation. Areas under analysis in the NSF study which will be included as test sites for this mission are the coasts of Oregon, Peru, and Somali. An important criterion for this study is a repetition of data acquisition from the test site area. Isolated data acquisitions are of little research value in studying these various phenomena.

Sea Surface Temperature. This mission will consist of proving measurement techniques in well instrumented areas where surface surveys are in progress. Concentrated measurements will be made in the Grand Banks area and in upwelling regions off the coasts of Spanish Sahara and Peru.

Geoid Measurement. Global low latitude ocean geoid measurements are desired in this mission. This data will supplement similar high inclination data obtained by GEOS-3 and Seasat-A, and provide a means for calibrating this satellite data. Since this flight involves an inclination of 57°, continuous data throughout several orbits is desirable.

Sea Ice Survey. Implicit in the title of this experiment is its association with ocean areas in the northern latitudes. Since this mission is being flown at an inclination of 57°, the existence of sea ice must be associated with seasonal conditions. The launch date for this flight is in August; consequently, the likelihood of sea ice at latitudes of 57° in the northern hemisphere is remote. However, sea ice is probable in the southern hemisphere at this latitude in August. The Cape Horn area at the southern tip of South America, therefore, becomes the only significant target area where sea ice might interfere with shipping during the time period of this flight.

Crustal Motions/Land Subsidence. Two key areas exist within the continental U.S. for the conduct of this experiment: the San Andreas fault region in California for crustal motion, and the southwestern corner of Flordia where the Everglades are being drained by land developers resulting in the subsiding of the land mass. This experiment will provide an excellent opportunity to develop sensors and techniques for future monitoring systems while providing an early baseline of data. The relationship of these target areas to the orbital ground tracks results in an opportunity for measurement approximately once a day; thus, ensuring a high probability of achieving substantial data on two very important target areas.

Ocean Currents. Ideally, this experiment should be concentrated on areas where internal wave patterns and strong western boundary currents are known to exist. The Sea of Japan and the Gulf Stream are two desirable target areas satisfying these conditions. Correlation

of these target areas with the orbital ground tracks indicates multiple opportunities for observing the Sea of Japan; however, the Gulf Stream is not traversed by the ground tracks. Consequently, the conduct of this experiment is only performed on passes over the Sea of Japan.

Adaptive Multibeam Communications. For this experiment ship-to-ship and ship-to-shore communication links via AMPA are planned for two broad ocean areas: an area approximately 500 nm southwest of San Diego, and another area approximately 400 nm east of Cape Hatteras. These areas offer frequent contact opportunities since they encompass areas with heavy ship traffic, and coincide with a crossover area for ascending and decending orbital ground tracks. Ship-to-shore links via AMPA will also be demonstrated from these areas to the STDN stations at Goldstone, California and GSFC, Maryland.

The land mobile demonstrations are planned to be performed with mobile vans operating out of Tinker AFB in Oklahoma and Glenview NAS in Illinois. Calibrations will also be obtained on other overflights of the Goldstone and GSFC STDN stations..

Ozone Mapping. While the total area of interest for this mission is global, those areas over continental land masses are of first priority. This prioritization occurs because the depletion of ozone is the primary concern; and the sources causing this depletion are primarily associated with activities on land. From a calibration standpoint, there is no preferred location. Correlation of orbital ground tracks with land mass crossings resulted in the selection of North America, Europe, Africa, and Asia as the major target areas for this mission.

Cloud Climatology. This mission is essentially independent of geographic area since the targets are clouds, which can be found almost globally. However, statistical probabilities indicate latitudes between 0° - 15° and 30° - 50° are most promising.

Table 2-11 details the various target locations and selected data taking operations while Table 2-12 summarizes this data. It is noted that while each mission/experiment is scheduled multiple times, individual target locations are frequently acquired only two or three times throughout the flight due to availability, visibility, or conflict with other experiments. This is particularly

FOLDOUT FRAME 1

Table 2-11. Mission Operations

Mission/Experiment	Site	Latitude	Longitude	Orbit	Mission Time (minutes)		Remarks
					Initiation	Termination	
Timber Inventory	Willamette National Forest	43.5 - 44.5	121.5 - 122.5	40	3472.0	3472.5	
				56	4887.0	4888.0	No SIR
Timber Inventory	Apache National Forest	33 - 34	108.3 - 109.4	8	645.0	645.5	
				40	3475.5	3476.5	
Timber Inventory	Green Mtn National Forest	42.7 - 44	72.7 - 73.3	86	7542.0	7542.5	No SIR
				102	8957.0	8957.5	
Timber Inventory	Talladega National Forest	33 - 33.7	85.7 - 86.3	55	4802.5	4803.0	
				71	6218.5	6219.0	No SIR
Timber Inventory	Clark National Forest	37.3 - 38	90.8 - 91.4	23	1970.5	1971.0	
				39	3386.0	3386.5	No SIR
Urban Planning	Miami, FL	25.7	80.3	39	3389.5	3390.0	
				55	4806.0	4805.5	
				71	6220.0	6220.5	
Urban Planning	New Haven, CT	41.2	72.5	38	3296.0	3296.5	
				54	4711.5	4712.0	
				70	6127.0	6127.5	
				86	7542.0	7542.5	
Urban Planning	Portland, OR	45.5	122.6	56	4887.0	4887.5	
				72	6302.5	6303.0	
				88	7717.5	7718.0	
				104	9133.0	9133.5	
Urban Planning	El Paso, TX	31.7	106.0	72	6307.0	6307.5	
				88	7722.5	7723.0	
				104	9138.0	9138.5	
Urban Planning	Birmingham, AL	33.3	86.5	55	4802.5	4803.0	
				71	6217.5	6218.0	
				87	7633.0	7633.5	
Mineral Exploration	Arizona	35.5 - 37	110 - 111.5	40	3475.0	3475.5	
				56	4890.0	4891.0	No SIR
				72	6305.5	6306.5	
				88	7720.5	7721.5	No SIR
Mineral Exploration	Nevada	38.5 - 40	114 - 115.5	19	1595.5	1596.5	
				40	3473.5	3474.5	
				56	4889.0	4890.0	No SIR
Mineral Exploration	Eastern Europe	48 - 60	60 - 70E	14	1165.0	1165.5	
				30	2580.5	2581.0	
				62	5411.0	5411.5	
				78	6826.5	6827.0	No SIR
				94	8241.5	8242.0	No SIR
Vegetation Stress	Iowa	40.7 - 42.5	93.5 - 95.0	55	4800.0	4800.5	
				71	6215.5	6216.0	No SIR
				87	7630.5	7631.5	
				103	9046.0	9046.5	No SIR
Marine Resources	Oregon Coast	46 - 47	124 - 126	40	3471.0	3472.0	
				68	5933.0	5934.0	
				72	6301.5	6302.5	
				84	7348.5	7349.0	
Marine Resources	Somali Coast	3	47	82	7200.0	7200.5	
				98	8615.5	8616.0	
Range Inventory	Central Florida	27.2 - 28	80.8 - 81.8	23	1973.5	1974.5	
				39	3389.0	3389.5	No SIR
				55	4804.5	4805.0	
Range Inventory	West Texas	31 - 32	104 - 105	88	7722.5	7723.0	No SIR
				104	9138.0	9138.5	
Range Inventory	Northern Nevada	41 - 42	116 - 117	56	4888.5	4889.0	No SIR
				72	6304.0	6304.5	
				88	7719.0	7720.0	No SIR
				104	9134.5	9135.0	
Water Inventory	Southern Florida	25 - 26.5	80.5 - 81.3	23	1974.0	1975.0	
Water Inventory	Tennessee	34.3 - 36.3	87.5 - 88.5	23	1971.5	1972.0	
				71	6217.0	6218.0	No SIR
				87	7632.5	7633.5	
Water Inventory	Arizona	33 - 34	109.5 - 111	8	644.0	645.5	No SIR
				40	3475.0	3476.0	No SIR
Water Inventory	Oregon	43.6 - 44.6	119.5 - 121	56	4887.5	4888.0	No SIR
				72	6303.0	6303.5	
				88	7718.0	7719.0	No SIR
				104	9133.5	9134.0	

FOLDOUT FRAME 2

Table 2-11. Mission Operations (Cont'd)

Mission/Experiment	Site	Latitude	Longitude	Orbit	Mission Time (minutes) Initiation	Termination	Remarks
Crop Survey	Alabama/Georgia	31 - 32	84.5 - 85.5	23	1972.5	1973.0	
				39	3387.5	3388.0	No SIR
Crop Survey	Southeast Missouri	36 - 37.6	89 - 90.5	23	1970.5	1973.0	
				39	3386.0	3388.0	No SIR
				55	4801.5	4802.0	
				71	6216.5	6217.5	No SIR
				87	7632.0	7633.0	
Crop Survey	Iowa	41	95.4	39	3384 0	3384.5	No SIR
				55	4799.5	4800.0	
Crop Survey	South Dakota	43	96 6	39	3383.5	3384.0	No SIR
				55	4799.0	4799.5	
Crop Survey	Eastern S. Dakota	44 - 45	96.3 - 97 5	87	7630.0	7630.5	
				103	9045.0	9045.5	No SIR
Crop Survey	Eastern China	30 - 31	119 - 120E	62	5423 0	5423.5	
				78	6838 0	6839.0	No SIR
				94	8253.5	8254.0	
Crop Survey	Western Russia	48 - 49	50 - 51E	32	2762.5	2763.0	
				48	4178.0	4178.5	
				64	5593.0	5594.0	
				80	7008.5	7009.0	
				96	8424.0	8424.5	
Crop Survey	Spain	38 - 39	1 - 2W	51	4447.0	4447.5	
				67	5862.5	5863.0	
				83	7277.5	7278.5	No SIR
				99	8693.0	8693.5	
Crop Survey	Western India	29.5 - 30.5	73 - 74	48	4184.5	4185.0	
				64	5599.5	5600.0	
				80	7015.0	7015.5	
				96	8430.0	8430.5	
Crustal Motion	Southern California	33 - 36.9	116 - 121.7	19	1594.5	1595.0	
				35	3010.0	3010.5	
				51	4425.0	4425.5	
				67	5840.5	5840.5	
				83	7256.0	7256.5	
				99	8671.0	8671.5	
Land Subsidence	Southern Florida	25.1 - 26.9	80.2 - 81.6	49	4245.5	4246 0	
				65	5660.5	5661.5	
				81	7076 0	7076.5	
Ocean Currents	Sea of Japan	34.2 - 44.1	127.6 - 142.1	45	3915.4	3917.0	
				61	5330.7	5332.4	
				77	6746.1	6747.7	
				93	8161.4	8163.1	
Sea Surface Temperature	Grand Banks	42.8 - 48.8	46.4 - 59.1	69	6036.2	6037.1	
				85	7451.5	7452.7	
				101	8866.9	8868.2	
Sea Surface Temperature	Peruvian Coast	0.2 - 15.4S	69.1 - 81.7	88	7733.3	7736.5	
				104	9148.7	9152.0	
Sea Surface Temperature	Spanish Sahara Coast	21.1 - 28.4	13.1 - 18 6	52	4538.8	4539.3	
				68	5954.3	5954.6	
				84	7369 8	7369.9	
Geoid Measurement	Global	-	-	11-13	875.0	1141.0	
Sea Ice Survey	Cape Horn	54 5 - 60S	50 - 90	10	851.0	854.0	
				27	2266.0	2269.0	
				43	3770.0	3776.0	
				59	5186.0	5191.0	
				75	6601.0	6607.0	
				91	8016.0	8022.0	
Storm Assessment	Indian Ocean	0 ± 25	40 - 100	9	780.0	795.0	
				16	1355.0	1370.0	
				57/58	5026.0	5041.0	

Table 2-11. Mission Operations (Cont'd)

Mission/Experiment	Site	Latitude	Longitude	Orbit	Mission Time (minutes)		Remarks
					Initiation	Termination	
Storm Assessment	Pacific Ocean	0 ± 25	139.5 - 173.5	26	2240.0	2255.0	
				42	3655.0	3670.0	
				74	6486.0	6501.0	
Storm Assessment	North Atlantic	0 - 25N	6 - 52.6	31	2645.0	2653.0	
				37	3213.0	3220.0	
Cloud Climatology	Global	-	-	15	1239.0	1249.0	
				20	1712.0	1221.0	
				21	1788.0	1796.0	
				24	2054.0	2062.0	
				28	2389.0	2396.0	
				36	3115.0	3123.0	
				43	3734.0	3742.0	
				46	3981.0	3989.0	
				47	4088.0	4096.0	
				53	4619.0	4627.0	
				60	5238.0	5246.0	
				63	5504.0	5511.0	
				76	6654.0	6661.0	
				79	6919.0	6927.0	
				92	8069.0	8076.0	
				95	8334.0	8342.0	
				100	8777.0	8784.0	
Ozone Measurement	Global	20 - 50	-	22	1869.5	1881.5	
				35	3009.5	3021.0	
				44	3804.0	3825.0	
				50	4332.0	4350.0	
				96	8415.0	8434.0	
Multibeam Communications	GSFC to Ship 1	30 - 43	64 - 79	17	1416.0	1420.0	
				33	2831.0	2835.0	
				49	4246.5	4250.5	
				65	5662.5	5666.0	
				81	7077.5	7081.0	
				97	8493.0	8496.5	
	Mobile 1 to GSFC	33 - 48	76 - 89	18	1508.5	1510.0	
				34	2923.5	2926.0	
				50	4339.0	4341.5	
				66	5754.0	5756.5	
				82	7169.5	7172.0	
				98	8584.5	8587.5	
	Mobile 2 to Mobile 1	32 - 44	86 - 100	18	1505.0	1508.5	
				34	2921.0	2923.5	
				50	4337.0	4339.0	
				66	5752.0	5754.0	
				82	7167.5	7169.5	
				98	8583.0	8584.5	
	Ship 3 to Ship 2	20 - 36	120 - 133	9	733.0	738.0	
				25	2148.0	2152.5	
				41	3563.5	3566.0	
				57	4978.5	4983.5	
				73	6394.5	6398.5	
				89	7809.5	7814.0	
				105	9225.0	9229.0	
	Ship 2 to Goldstone	28 - 38	114.5 - 125	19	1592.0	1596.0	
				35	3007.0	3011.0	
				51	4423.0	4426.5	
				67	5838.5	5841.5	
				83	7253.5	7257.0	
				99	8669.0	8672.5	

Table 2-12. Mission Summary

Experiment	Number of Operations	Total Minutes of Data
Timber Inventory	10	7.0
Urban Planning	17	8.5
Mineral Exploration	12	9.0
Vegetation Stress	4	2.5
Marine Resources	2	1.0
Range Inventory	9	5.5
Water Inventory	10	8.5
Crop Survey	29	20.5
Crustal Motion/Land Subsidence	9	5.5
Ocean Currents	4	6.6
Sea Surface Temperature	8	10.8
Geoid Measurement	1	266.0
Sea Ice Survey	6	29.0
Storm Assessment	8	105.0
Cloud Climatology	17	134.0
Ozone Measurement	5	82.5
Multibeam Communications	31	120.5

true for the Earth Resource missions - partly due to the number of targets specified, and some times due to power constraints involving the use of the Shuttle Imaging Radar. While the use of this sensor is desired for most of the Earth Resources and Earth and Ocean Dynamics missions, it is not feasible to timeline its operation into every data taking opportunity. Consequently, as indicated under the remarks column, the SIR is not operated for every pass over every target desiring it; but is operated at least once over each such target. This is acceptable to the investigators since the microwave signature return is essentially a non-varying quantity as opposed to optical signatures for vegetation, sea surface, etc.

The assignment of orbital data taking segments was predicated on obtaining sufficient data over those targets observed only a few times as a first priority. (It should be noted here that the first and final eight orbits were arbitrarily excluded from any data taking and reserved for STS operations.) Orbital passes over CONUS were divided among the experiments based on geographical proximity of the target to the ground track, lighting, and number of opportunities. Finally, those experiments involving global or large area coverage were accommodated as fillers in the timelines because of their flexibility. The Storm Assessment, Cloud Climatology, and Ozone Mapping experiments are included in this category for the following reasons:

1. Storm Assessment. Global ocean areas, with the exception of the South Atlantic (low probability of storm origin), are of interest. Areas between $\pm 25^{o}$ latitude are of most interest.

2. Cloud Climatology. Essentially independent of geographic area since the targets are clouds, which can be found almost globally; however, statistical probabilities indicate latitudes between $0-15^{o}$ and $30-50^{o}$ are most promising.

3. Ozone Mapping. Total earth coverage is the ultimate goal; however, continental areas between $20 - 50^{o}$ North and South latitude are first priorities.

A definite attempt was made to achieve a patterned coverage over the entirety of these areas.

Snapshot illustrations for each experiment requiring definitive target coverage are shown in Figures 2-21 through 2-27.

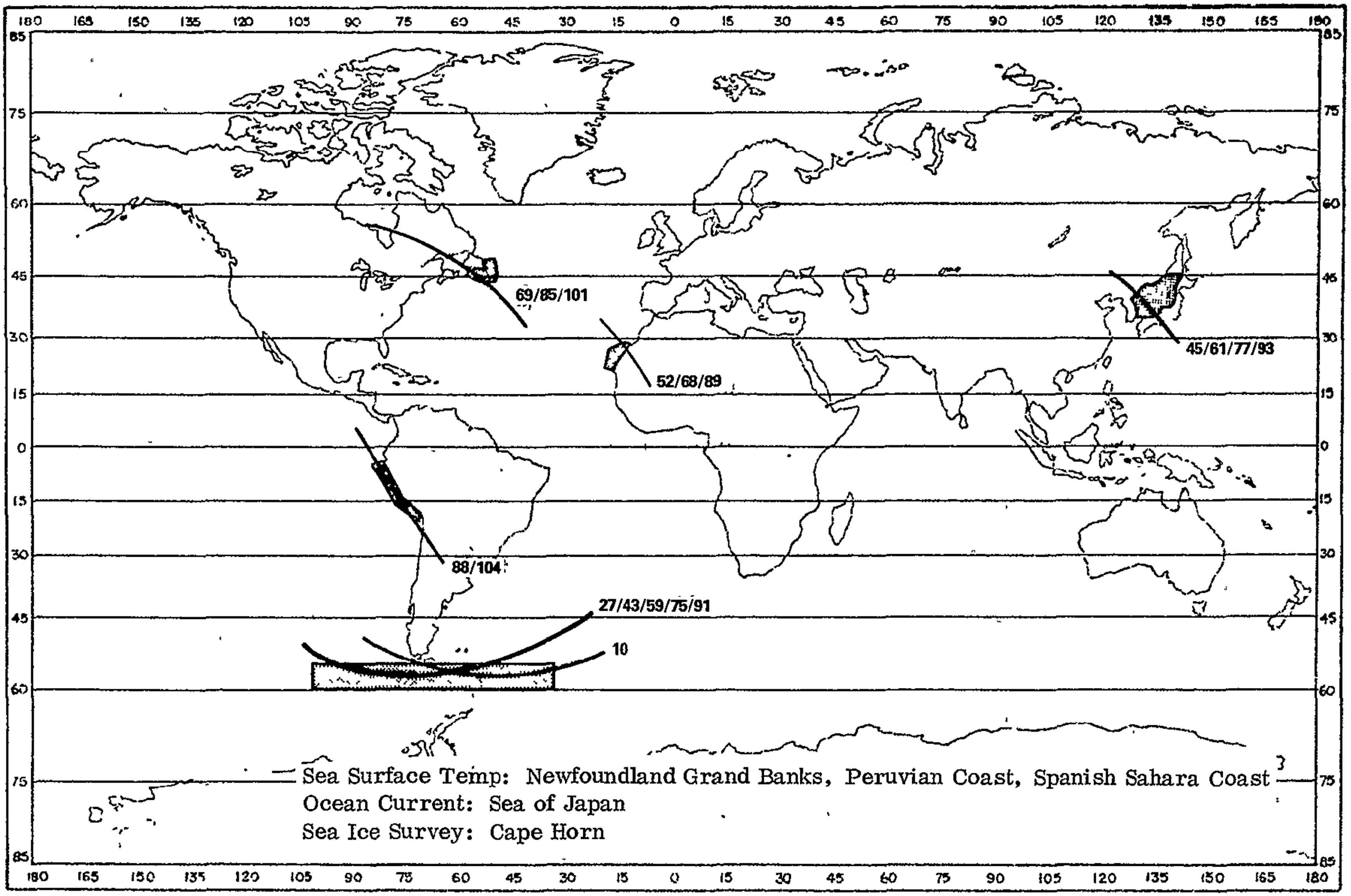


Figure 2-21. Ocean Areas

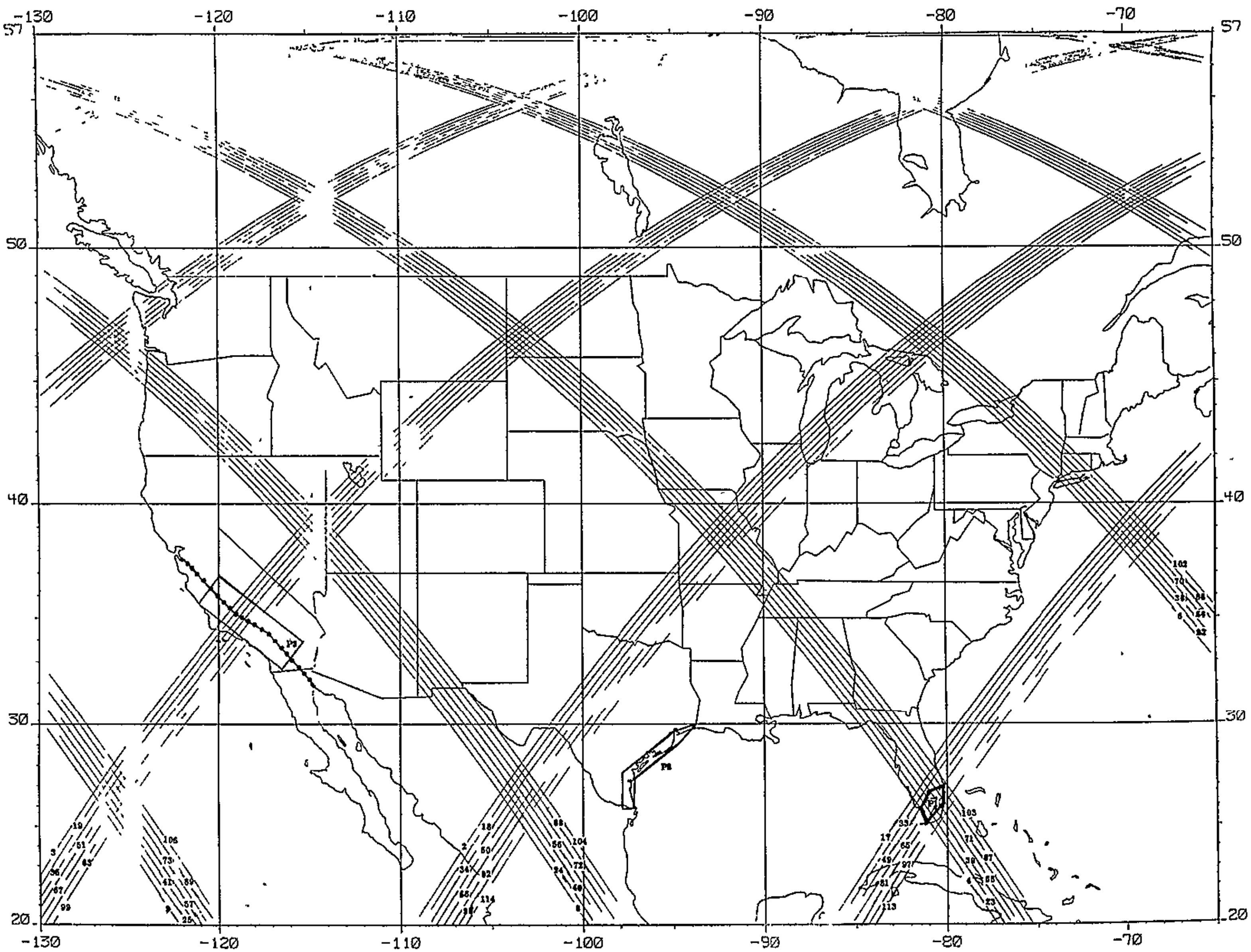


Figure 2-22. Crustal Motion/ Land Subsidence Sites

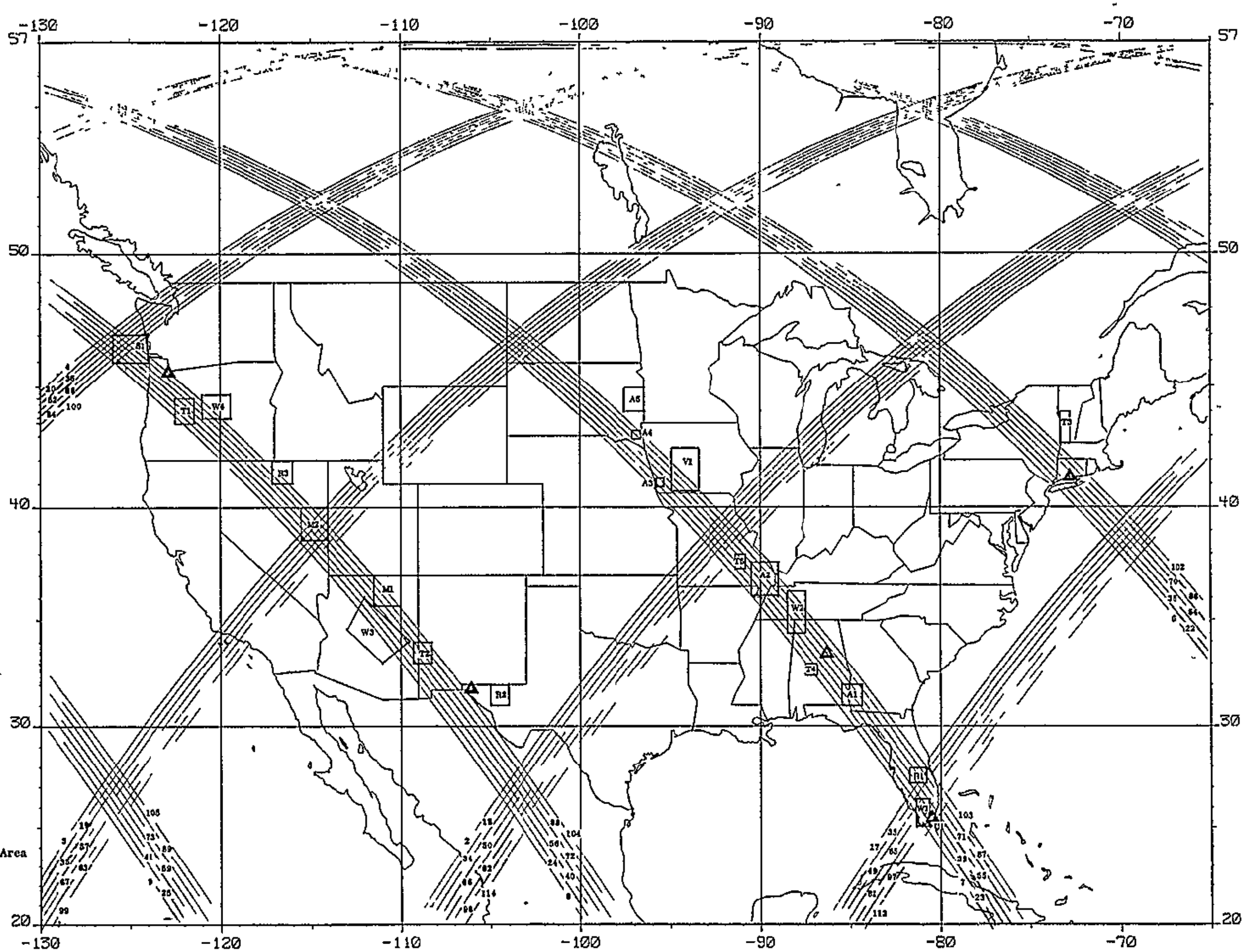


Figure 2-23. Earth Resources Sites

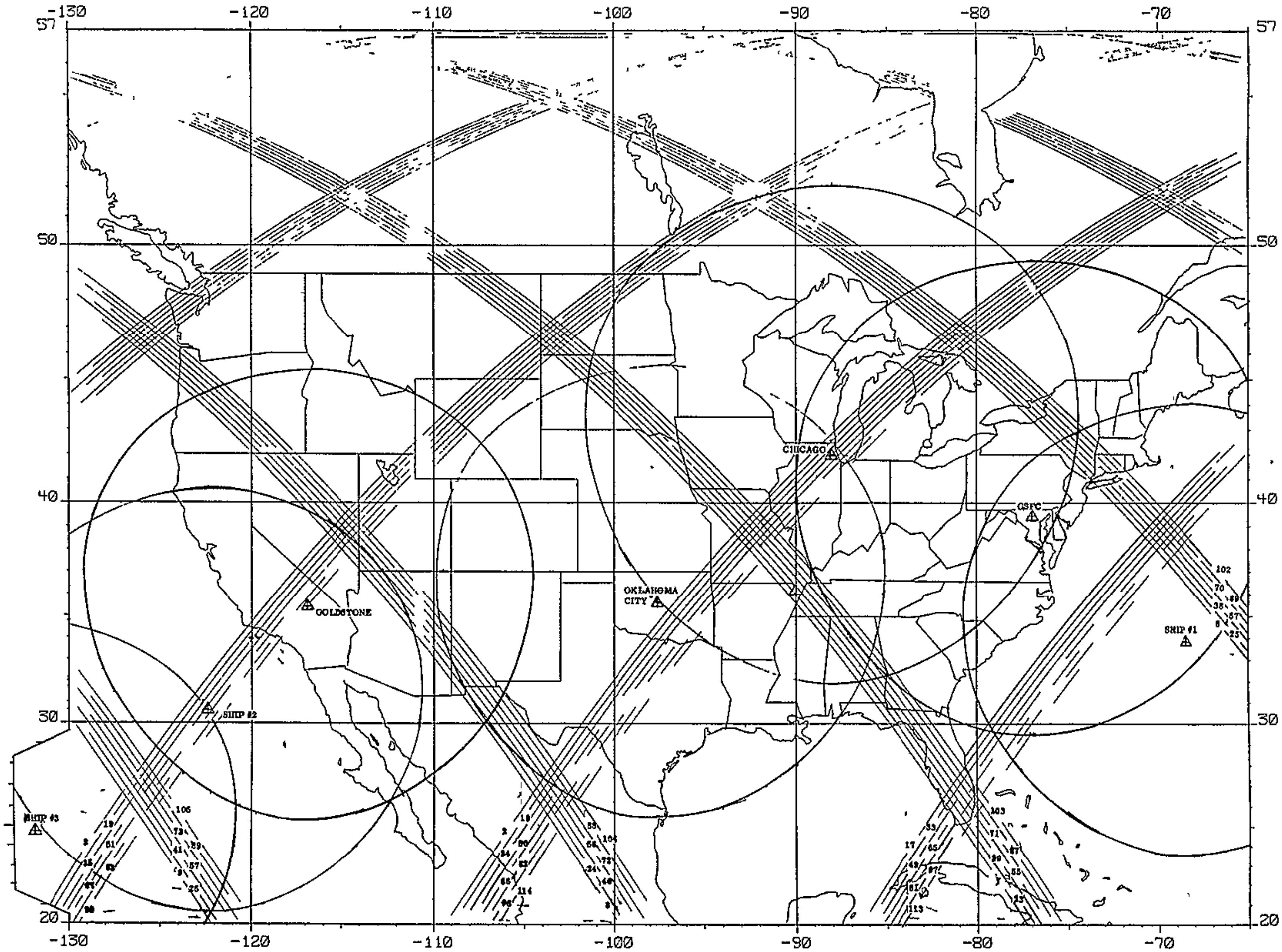


Figure 2-24 Multi Beam Communications Sites

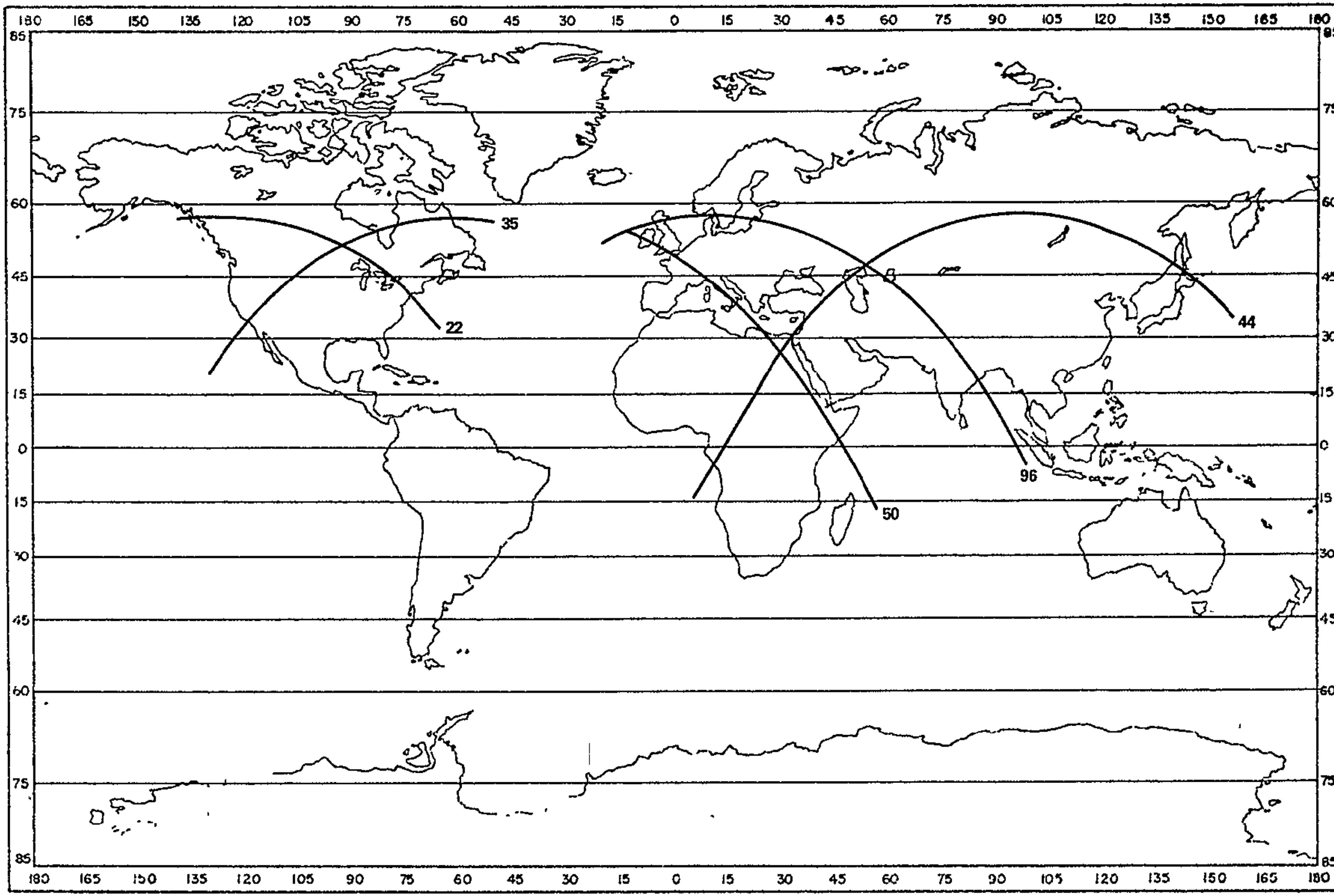


Figure 2-25. Ozone Mapping Regions

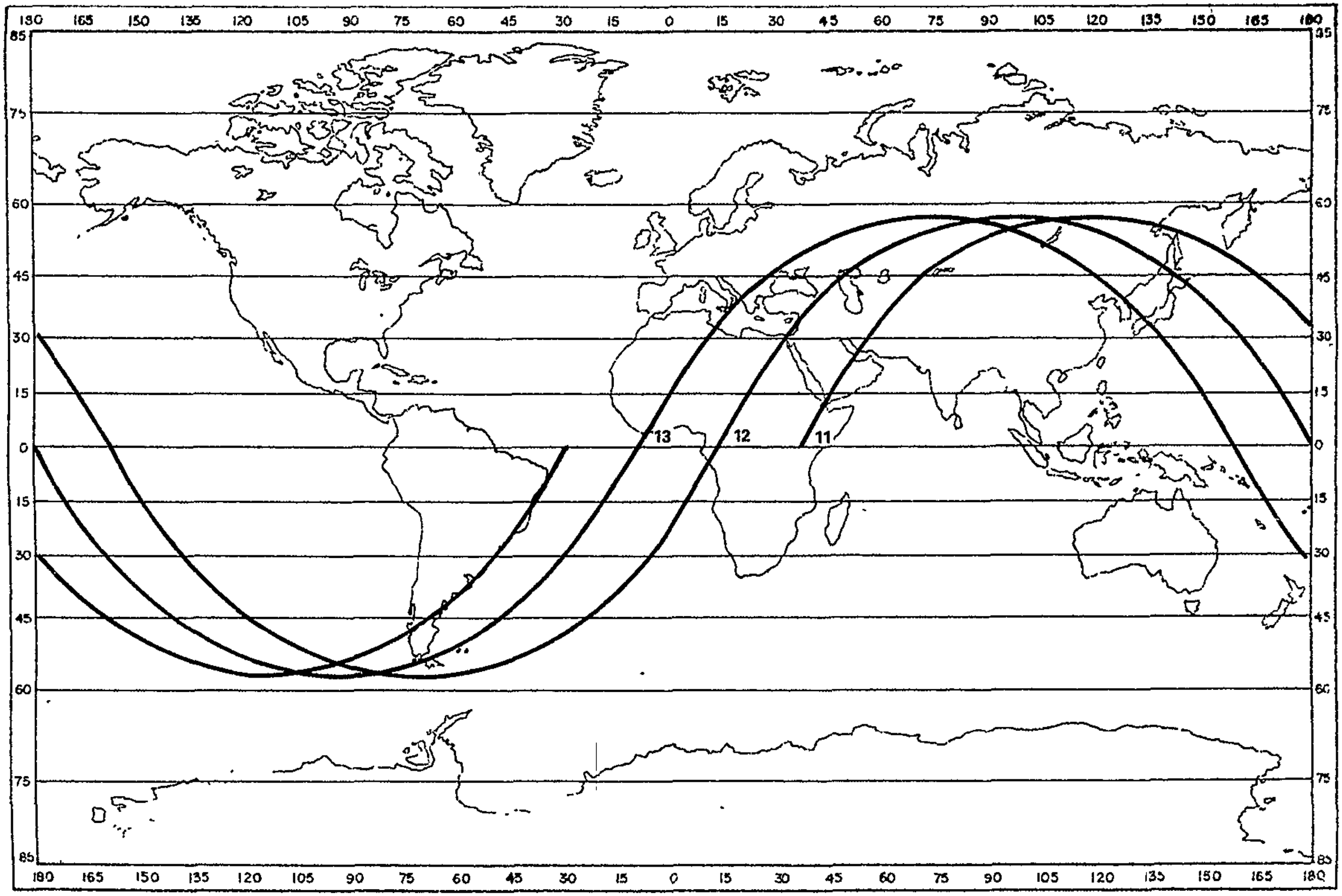


Figure 2-26. GEOID Measurement Area (Orbit 11/12/13)

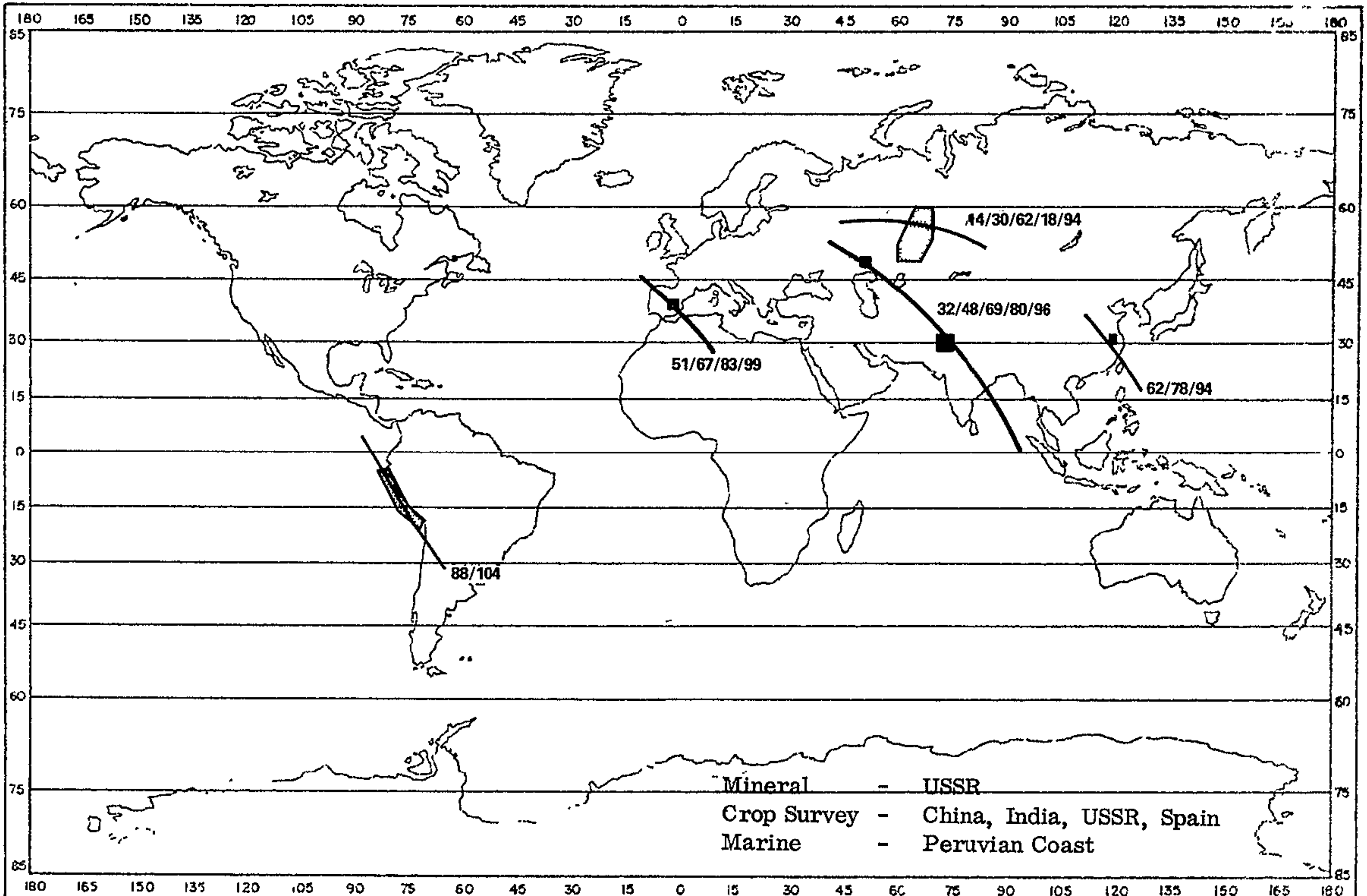


Figure 2-27. World Resources Sites

Also considered in the assignment of data taking opportunities for the various experiments was the probability of experiment success as it is influenced by cloud cover. Experiments such as Urban Planning, Crop Survey, Vegetation Stress, Rangeland Status, Mineral Survey, and Timber Inventory are dependent upon the ability to acquire good photographic data. For the purposes of this study, photographable skies are defined as those skies in which there is at least 75% visibility (up to 25% obscurity by haze or partly cloudly skies may exist). Information obtained from a reference document -"Further Developments in Cloud Statistics--," NAS CR-61389 - indicates the probability of clear and photographable conditions for various geographic locations. Table 2-13 summarizes this data for the period between August and September for the geographic areas of interest.

Table 2-13. Single Pass Probabilities for Cloud Conditions

	Clear	Photographable
Northeastern CONUS	28	45
Northwestern CONUS	45	64
Southwestern CONUS	41	70
Southeastern CONUS	23	43
Central CONUS	21	38
Eastern Europe	22	46
Eastern China	24	52
Western India	42	78
Spain	40	62

It is observed from Table 2-13 that there is a considerable range in the probability of encountering photographable conditions over the target areas of interest on a single pass. The real significance, however, is the combination of these probabilities and the number of opportunities which are available. This can be translated into a probability of mission/experiment success. Table 2-14 tabulates this data for each mission experiment. While there are relatively few opportunities for data acquisition on many targets, probability of mission success is generally quite high. The one mission not having a high probability of success is Marine Resources. This situation results principally from the high probability of cloud cover over the target areas

Table 2-14. Probability of Mission Success

	Probability of Photographable Conditions (%)	Number of Passes	Probability of Mission Success (%)
Timber Inventory			
Williamette, N.F.	64	2	87
Apache, N.F.	70	2	91
Green Mountain, N.F.	45	2	70
Talladega, N.F.	43	2	67
Clark, N.F.	43	2	67
Urban Planning			
Miami, Fl	25	3	58
New Haven, Conn	45	4	91
Portland, Ore	64	4	98
El Paso, Tex	70	3	97
Birmingham, Ala.	43	3	81
Mineral Exploration			
Arizona	70	4	99
Nevada	70	3	97
Eastern Europe	46	5	95
Vegetation Stress			
Iowa	52	4	95
Marine Resources			
Oregon Coast	22	4	61
Somali Coast	27	2	47
Range Inventory			
Central Florida	25	3	58
West Texas	67	2	89
North Nevada	64	4	98
Water Inventory			
Florida	25	1	25
Tennessee	43	3	81
Arizona	70	2	91
Oregon	64	4	98
Crop Survey			
Alabama/Georgia	43	2	67
Missouri	55	5	98
Iowa	52	2	77
Dakota	45	4	91
Eastern China	52	3	89
Western Russia	30	5	83
Spain	62	4	98
Western India	78	4	99

(the Oregon and Somali coasts). Similarily, the Florida sites for the Range Inventory, Water Inventory, and Urban Planning missions have only moderate probability of being successfully observed. Other test sites associated with these missions, however, offer high probabilities of mission success.

The missions involving foreign test sites all have high probabilities of success in large part due to the fact that they are geographically dispersed and have little competition for viewing time. Consequently, the number of available/selected data passes is higher and the probability of mission success is increased.

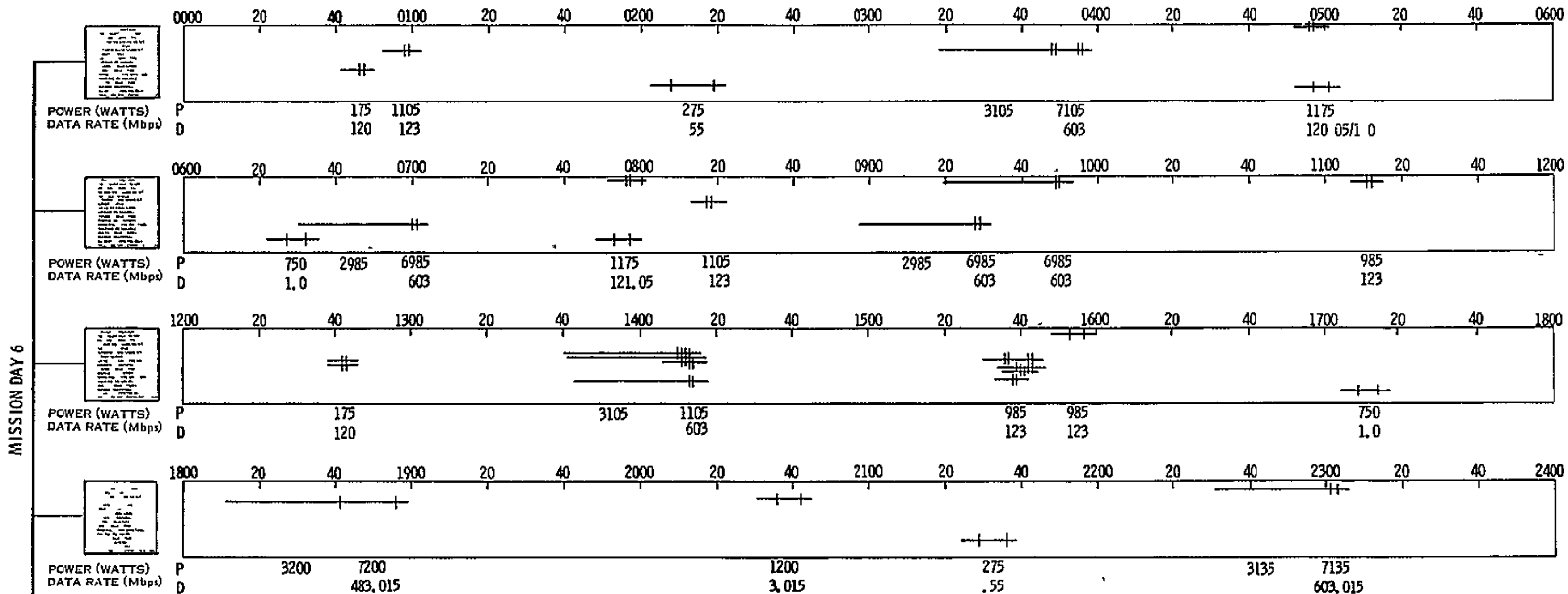
For the larger geographical target areas, the probability of mission success can probably be improved by some form of adaptive cloud avoidance. Optical and/or microwave systems could be developed to look ahead and discern cloud free areas to which the Orbiter could be maneuvered; or the system might be as simple as using a crewman to visually look ahead and select the most promising areas. Real-time coordination with observers physically located in the target areas might also prove feasible.

Mission Timelines

In addition to satisfying the previously mentioned experiment observation requirements, experiments were also timelined to achieve synergistic benefits whenever possible.

The overall process involved several iterations, with the resultant being a set of mission timelines. A sample of this mission timeline is shown in Figure 2-28. The power and data profiles shown across the bottom of these timelines indicate the power and data profiles for the EVAL sensors only. The mission "on" times are indicated by the horizontal dark lines, while the interval encompassed by the vertical tick marks on these lines denotes the actual data gathering period.

It is observed from the EVAL timelines that throughout the mission, operations are characterized by periods of high activity, followed by approximately one hour of no observations or measurements, and then another period of activity followed by another period of inactivity.



1. CRUSTAL MOTION
2. SEA SURFACE TEMP.
3. OCEAN CURRENTS
4. GEOID MEASUREMENT
5. SEA ICE SURVEY
6. STORM ASSESSMENT
7. CROP SURVEY
8. VEGETATION STRESS
9. URBAN PLANNING
10. TIMBER INVENTORY
11. RANGE INVENTORY
12. MINERAL EXPLORATION
13. MARINE RESOURCES
14. WATER INVENTORY
15. OZONE MAPPING
16. CLOUD CLIMATOLOGY
17. MULTIBEAM COMMUNICATION

Figure 2-28. Mission Timeline

This cycle is essentially repeated throughout the flight, and is caused by a combination of factors involving lighting and geographical locations. Because of the launch conditions chosen, the southern hemisphere and India/Asia/China are generally overflown during periods of darkness. This lack of lighting, coupled with the fact that few experiment target areas are located over these areas, accounts for the cyclical periods of inactivity. This characteristic is highly desirable in that it allows the Shuttle/Spacelab crew, as well as the principle investigators on the ground, time to briefly evaluate the just-acquired data and plan for the next data take.

Another characteristic of these timelines is the frequent overlapping or consecutive occurrance of multiple missions in a short time span - as exemplified by the mission activity around times 1415 and 1540 of day 6 shown in Figure 2-28. These intervals represent passes over CONUS where prime mission test sites are located. Since there are a limited number of such passes, and the number of test sites is large, this type of operation is necessary. With a single exception, there is no problem with this type of operation - in fact, there are appreciable advantages in that the sensors and crew can remain on active standby during data takes and avoid multiple, lengthy start-up and shut-down times. This results in a net power savings and mission continuity for the crew. The one drawback is that occasionally a sensor is required for two missions on opposite sides of the ground track, and cannot be slewed from one target to the other fast enough. In this case, the sensor in question is assigned to one of the targets on the initial pass, and the other target on a subsequent pass. Two examples of this type of conflict are shown in the expansion of the day 6, 1415 and 1540 time periods, operation shown in Figures 2-29 and 2-30. Figure 2-29 show the conflict between the Urban Planning mission El Paso, Texas site and the West Texas Range Inventory mission; while Figure 2-30 involves a conflict between the Tennessee Water Inventory mission and a Crop Survey test site in S.E. Missouri. The sensor in conflict in these instances is the Thematic Mapper. Figures 2-29 and 2-30 also give an indication of the diversity of missions being accomplished during this brief time period.

Figure 2-29. Expanded Mission Timeline
(1410 - 1414)

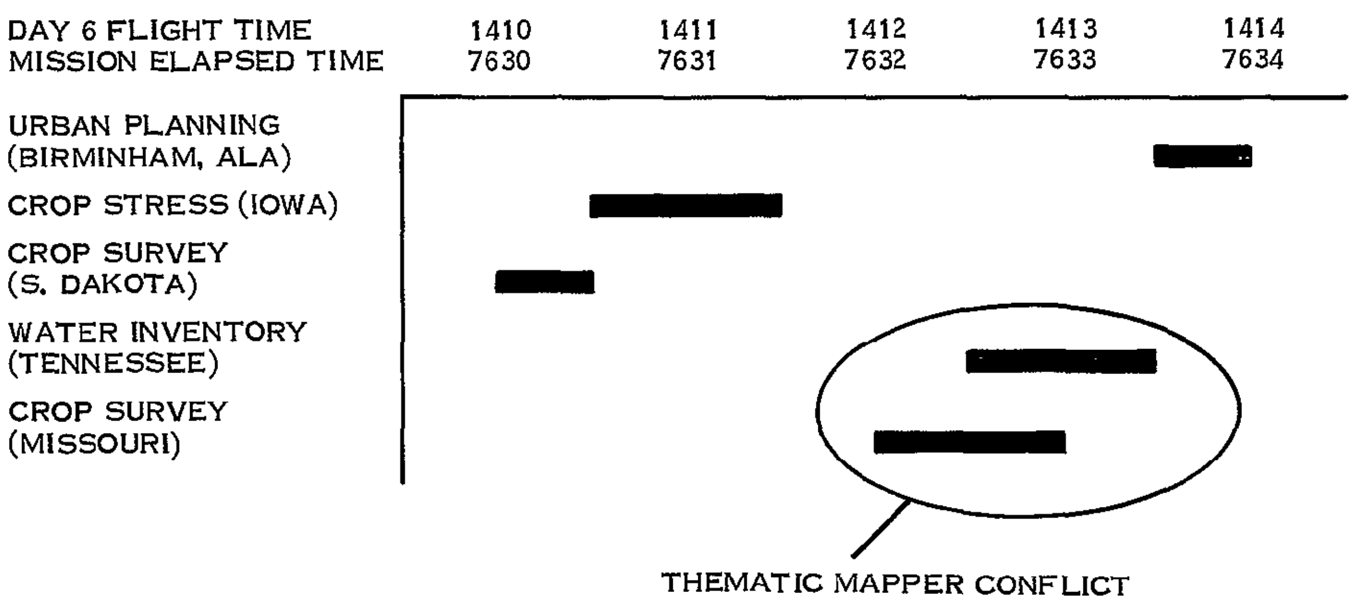


Figure 2-30. Expanded Mission Timeline
(1537 - 1543)

The time frame around 1415 and 1540 of day 6 shown in Figure 2-30 also exemplifies another facet of mission operations on this flight. In this instance the item of concern is power. Several of the missions identified in the expansions for these time periods (Figures 2-29 and Figure 2-30) require use of the Shuttle Imaging Radar. However, the SIR involves a lengthy (30 minute) warming at 2 kW and operates at 6 kW during data taking operations. This high power requirement necessitates drawing on the peak power account of Spacelab. While this is permissible, there are rules associated with such operations. In particular, payloads are limited to such operations for only 15 minutes every 3 hours. In this particular circumstance, the elapsed time is only approximately 1 hour and 15 minutes. Consequently, SIR operation is not feasible for one of the data passes - SIR was selected to be operated during the 1415 time period but not the 1540 period in this instance. It should be noted that this particular combination of missions and targets appears several times throughout the flight, and the assignment of SIR is varied to assure that each target is observed by SIR at least once.

Crew Requirements

Crew requirements emanating from the mission timelines indicate a two shift on-orbit operation. The Cloud Climatology experiment is conducted intermittently throughout the flight and requires a high degree of training and on-orbit dedicated operation; therefore, two payload specialists are required to operate this experiment and be responsible for the majority of the other experiments. Because of simultaneous experiment operations, Orbiter crew support was also utilized for monitoring selected payload experiments. (It is assumed that the Orbiter mission specialist would be the primary crew member assisting in the experiments, with additional support provided by either the commander or co-pilot, as available.) The total number of personnel required on-board for this flight, therefore, is five; commander, co-pilot, mission specialist, and two payload specialists. Crew operational assignments were developed under the following ground rules:

1. Each work day contains an eight hour sleep period where possible.
2. A minimum of six hours of sleep is required by all crewmen prior to re-entry.
3. Three hours of each workday is required for the three meal periods.

4. 1-1/4 hours of each work day is allocated to crew pre- and post-sleep activities (PSA).

5. 1-1/2 hours of each work day is allocated to crew planning and shift change activities.

6. Payload Specialists are the prime operators of payload equipment with Orbiter crew support as required.

7. The first and last eight orbits are dedicated to Orbiter/Spacelab activation functions.

8. Payload experiment operations are terminated at 18:00 hours of day seven (Orbit 104)

Figure 2-31 shows a typical timeline for a particular day.

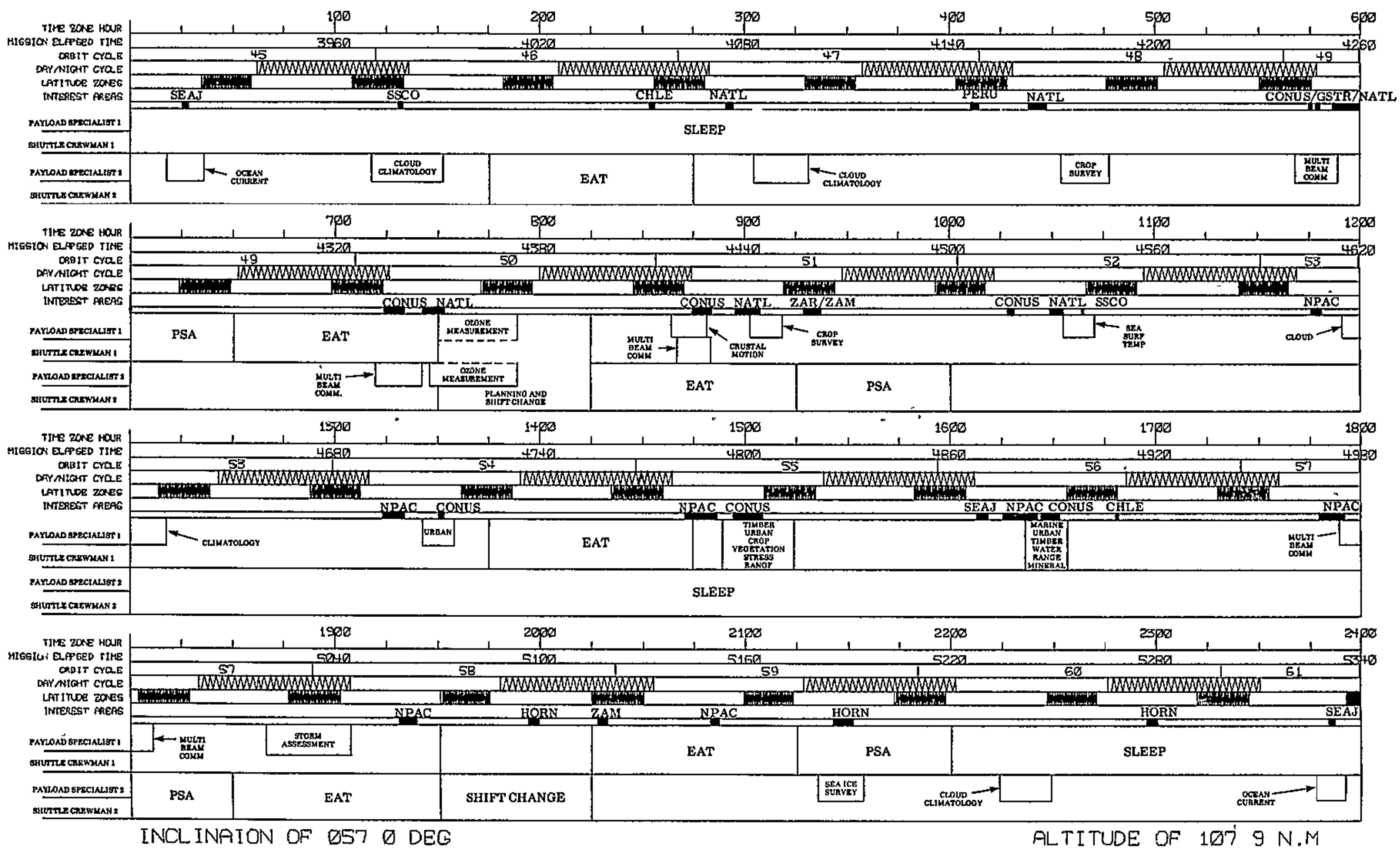


Figure 2-31. Mission Day 4

2.4.2 GROUND OPERATIONS

Physical integration of Shuttle payloads prior to flight will involve four levels of activity, beginning with checkout of payload subassemblies (Level IV) and ending with installation of cargo into the Orbiter (Level I). The precise definition of each integration level has shown some amount of evolution as Shuttle era concepts have matured. As of December 1976, the levels of integration as they apply to EVAL payloads are as follows:

1. Level IV. Integration and checkout of EVAL equipment assemblies with individual racks or pallet segments, or with the STR. Two examples are mounting the SIR and AMPA on a Spacelab pallet segment, and installing the cloud climatology control electronics into a Spacelab rack. Note that in many cases a major precursor effort is required to configure, assemble, checkout, and generally prepare the experiment equipment for Level IV integration.

2. Level III. Combination, integration, and checkout of all experiment/payload mounting elements (e.g., Spacelab racks and pallet segments, STR bridge pallet) with experiment/payload equipment already installed; and of experiment/payload and Spacelab software. Here Spacelab racks and/or pallet segments containing EVAL and other payload equipment are assembled into their flight configuration and checked out as a system. STR is included in this checkout when it is an integral part of the Spacelab payload.

3. Level II. Integration and checkout of the combined experiment/payload equipment and their mounting elements (e.g., Spacelab racks and/or pallet segments, STR bridge pallet) with flight support elements (e.g., Spacelab module segments or igloo).

4. Level I. Integration and checkout of total cargo (Spacelab modules and/or pallets and STR plus any other payloads such as automated spacecraft or piggyback packages) with the Shuttle Orbiter, including the necessary pre-installation assembly and testing with simulated interfaces.

The current baseline is that Levels I, II, and III integration will be at KSC while Level IV will take place at other sites. For EVAL, Level IV integration probably will be performed at the site of an integration contractor along with most pre-Level IV assembly and test of experiment equipment. Some payloads in combined EVAL missions may be integrated elsewhere in the USA or abroad, and need not meet the EVAL payloads until both arrive at KSC for Level III integration. The precise interface between Level IV and Level III integration is presently under re-evaluation and is consequently unclear. For this

study Level IV integration is assumed to end with validation with individual Spacelab racks and pallet segments and the STR bridge.

Preflight integration activities are summarized in Figure 2-32. Principal Investigators and/or their teams will be expected to participate in all levels of physical integration, actively supporting EVAL equipment installation, integration, and performance testing. A key PI activity is participation in mission simulation exercises, where on-orbit and ground support personnel rehearse their respective roles. The PI and/or his team will also support payload/cargo integration and servicing activities at KSC.

Post-flight activities begin as soon as the Orbiter has landed and has been towed to the Orbiter Processing Facility. The payload bay doors open approximately 15 hours after landing and Spacelab removal is completed at 30 hours. The Spacelab is then transported to the O&C Building for disassembly. Racks and pallets are demated from the Spacelab module and returned data is recovered. The PI and/or his team will support payload de-integration and data strip-out. Racks and pallets may be de-integrated at KSC and the experiment shipped north, or the racks and pallets themselves may be shipped north for de-integration. Factors such as rack and pallet ownership, re-use demand, and experiment transportability will be taken into account. Once in receipt of his data and equipment, the PI is free to use it or dispose of it as he sees fit.

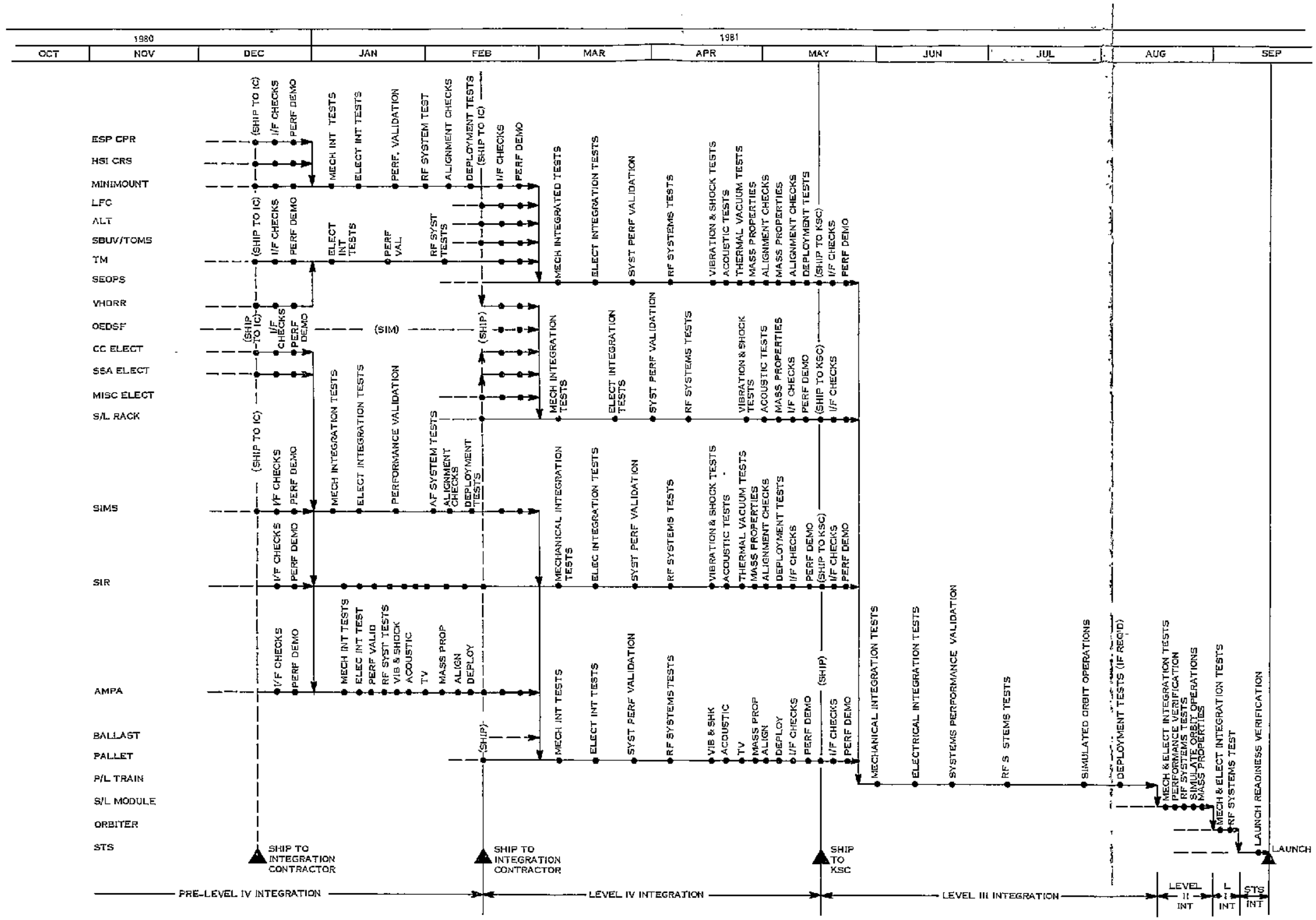


Figure 2-32. Physical Integration Activities for Dedicated EVAL Payload

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2.5 SUBSYSTEM SUPPORT

2.5.1 DATA MANAGEMENT

The sensors associated with this payload span a broad range of data rates, varying from 320 bps to 480 x 10^6 bps. Complicating the situation still further is the fact that several of these sensors (generally the ones with higher data rates) are required to operate simultaneously for many of the experiments/missions. This quite frequently results in experiment data rates on the order of 600 x 10^6 bps. Table 2-15 illustrates this condition.

The chief contributors to these high experiment data rates are the thematic mapper and the Shuttle Imaging Radar (a Synthetic Aperture Radar). Examination of the mission timelines indicates that these sensors are operated either separately or in combination for appreciable periods of time. This information is summarized in Table 2-16. The standard and optional facilities for data handling on-board the Shuttle/Spacelab cannot record at data rates in excess of 32 Mbps or transmit at rates over 50 Mbps; consequently, alternative techniques must be developed.

The solution to the TM data handling problem was discussed in detail in the EVAL Partial Spacelab Payload Technical Report, 76SDS4269, dated 30 September 1976. The conclusion was that an adaptation of a very high data rate recorder capable of handling a 120 Mbps data stream with 36 minute recording time would be the most economical solution. This conclusion is equally applicable to the TM data associated with the dedicated EVAL flight. A candidate recorder (HRDM-240S) is currently being developed by RCA for NASA/GSFC. This device is capable of recording two channels of 120 Mbps data simultaneously, with total record time of up to 18 minutes. Pertinent specifications are:

Volume	5.3 ft^3
Power	270 Watts
Weight	250 lbs (estimated)

It is estimated that this recorder will be available in the 1980-1981 time frame.

Table 2-15. Sensor Data Rates and Mission Utilization for Dedicated EVAL

Sensor	Acronym	Nominal Data Rate (Bps)	Crustal Motions	Sea Surface Temperature	Ocean Currents	Geoid Measurements	Sea Ice Survey	Storm Assessment	Crop Survey	Vegetation Stress	Urban Planning	Timber Inventory	Range Inventory	Mineral Exploration	Marine Resources	Water Inventory	Ozone Mapping	Cloud Climatology	Multibeam Communications
Shuttle Imaging Microwave System	SIMS	3×10^6		R	R		R	R	R	R			D		D	D			
Shuttle Imaging Radar	SIR/SAR	480×10^6			R		R	R	R	R		R	R	R	R	R			
Thematic Mapper	TM	120×10^6	D	D	R				R	R	R	R	R	R	R				
Large Format Camera	LFC	N/A	D				D	D	R	R	R	R		R					
GOES-C Altimeter		15×10^3			R	R	D	R											
Cloud Physics Radiometer	CPR	500×10^3																R	
Spaceborne Laser Ranging System	LRS	50×10^3	R															R	
Solar Backscatter UV Spectrometer/Total Ozone Mapper	SBUV/TOMS	320															R		
Adaptive Multibeam Phased Array	AMPA	1×10^6																	R
Total Data Rate	Desired	$604\ 6 \times 10^6$	50 K	3M	600 15M	15 K	483 M	483 05 M	603 M		120 M	600 M	600 M	600 M	600 M	480 M	320	550 K	1 M
	Tot Desired		120.5 M	123 M	603 15 M		483 015 M	483.05 M					603 M		603 M	483 M			

R Required

D Desired

Table 2-16. Usage of High Data Rate Sensors

	Missions	Duration (min.)
Simultaneous Usage of TM and SIR (600×10^6 Bps)	Ocean Currents	23
	Crop Survey	24
	Vegetation Stress	12
	Timber Inventory	11
	Range Inventory	10
	Mineral Exploration	11
	Marine Resources	6
	Total	97 minutes
Usage of SIR Without TM (480×10^6 Bps)	Sea Ice Survey	31
	Storm Assessment	109
	Water Inventory	7
	Total	147 minutes
Usage of TM Without SIR (120×10^6 Bps)	Crustal Motion	10
	Sea Surface Temperature	12
	Urban Planning	10
	Total	32 minutes

The data management of the SIR/SAR proves to be an even more complex task than for the thematic mapper. The Jet Propulsion Laboratory, in Report No. 750-73, Shuttle Synthetic Aperture Radar Implementation Study, dated March 8, 1976, states that the radar will have a data rate of about 120 Mbps for one frequency, at one polarization (at one look angle). (Independent evaluation by GE indicates that this data rate may be as high as 140 Mbps.) For multiple polarizations, look angles, and frequencies, the raw data rate can be expressed as:

$$f_R = f_D \times L \times N \times P \qquad (1)$$

where

f_R = raw data rate (bit/sec)

f_D = single channel data rate (120 - 140 Mbps)

L = number of look angles

N = number of frequencies used

P = number of polarizations used

The JPL report further states that it may be desirable to have at least two frequencies, two polarization, and from one to eight look angles. Applying these numbers to equation (1) result in:

$$f_R = f_D \times L \times N \times P$$

$$f_R = (120 \text{ Mbps}) \times (1) \times (2) \times (2) = 480 \text{ Mbps minimum}$$

or

$$f_R = (120 \text{ Mbps}) \times (8) \times (2) \times (2) = 3840 \text{ Mbps maximum}$$

These raw data rates are higher than those compatible with any equipment that is anticipated to be available by the 1980-81 period. It is conceivable that several HRDM-240S digital recorders could cope with these data rates. For example, two HRDM-240S could handle the minimum case (480 Mbps) but the extrapolation of this approach to the maximum case (3840 Mbps) results in 16 recorders. It should be noted that these figures involve raw, unprocessed data. On-board (digital) processing of the data can result in substantial reduction of the data rates, as shown below:

$$f_P = \frac{f_R}{L \times N \times P \times 2 \times F_I} \qquad (2)$$

where

f_P = Processed Data Rate (bits/sec)

f_R = Raw Data Rate (bits/sec)

L = Number of Look Angles

N = Number of Frequencies

P = Number of Polarizations

F_I = Integration Factor $\approx$ 5

f_D = Single Channel Data Rate = 120 Mbps

Thus:

$$f_P = \frac{f_D \times L \times N \times P}{L \times N \times P \times 2 \times F_I} = \frac{f_D}{(2) \times (5)} = \frac{f_D}{10}$$

or

$$f_P = \frac{120 \text{ Mbps}}{10} = 12 \text{ Mbps}$$

This is a manageable data rate compatible with the standard High Rate Digital Recorder (HRDR) associated with Spacelab flights.

Dependent upon the number of variables desired in the radar data, two solutions are proposed for handling the data associated with this payload. If a minimum SIR (one look angle) is assumed, its data can be routed to two HRDM-240S tape recorders operated semi-automatically in a man-rated environment. Data from the TM is accumulated on a separate HRDM-240S. The SIMS and AMPA data is routed to the experiment recorder and the other sensors are handled through the Shuttle orbiter data bus. This scheme is shown in Figure 2-33. In this method, SIR and TM data is returned on tape; all other data can be recorded and either stored, or, on option, be transmitted via a TDRS link.

For a higher order SIR, the solution assumes the availability of on-board processing for the SIR data resulting in a 12 Mbps data rate (not constraining on frequencies, polarizations or look angles). The SIR data, along with that from the SIMS and AMPA, is routed to the experiment recorder. The TM data is recorded on a modified HRDM-240S recorder and all other sensor data is routed through the Orbiter data bus. This scheme is shown in Figure 2-34 and is the preferred solution. The option to transmit all sensor data, except TM data, via TDRSS is available.

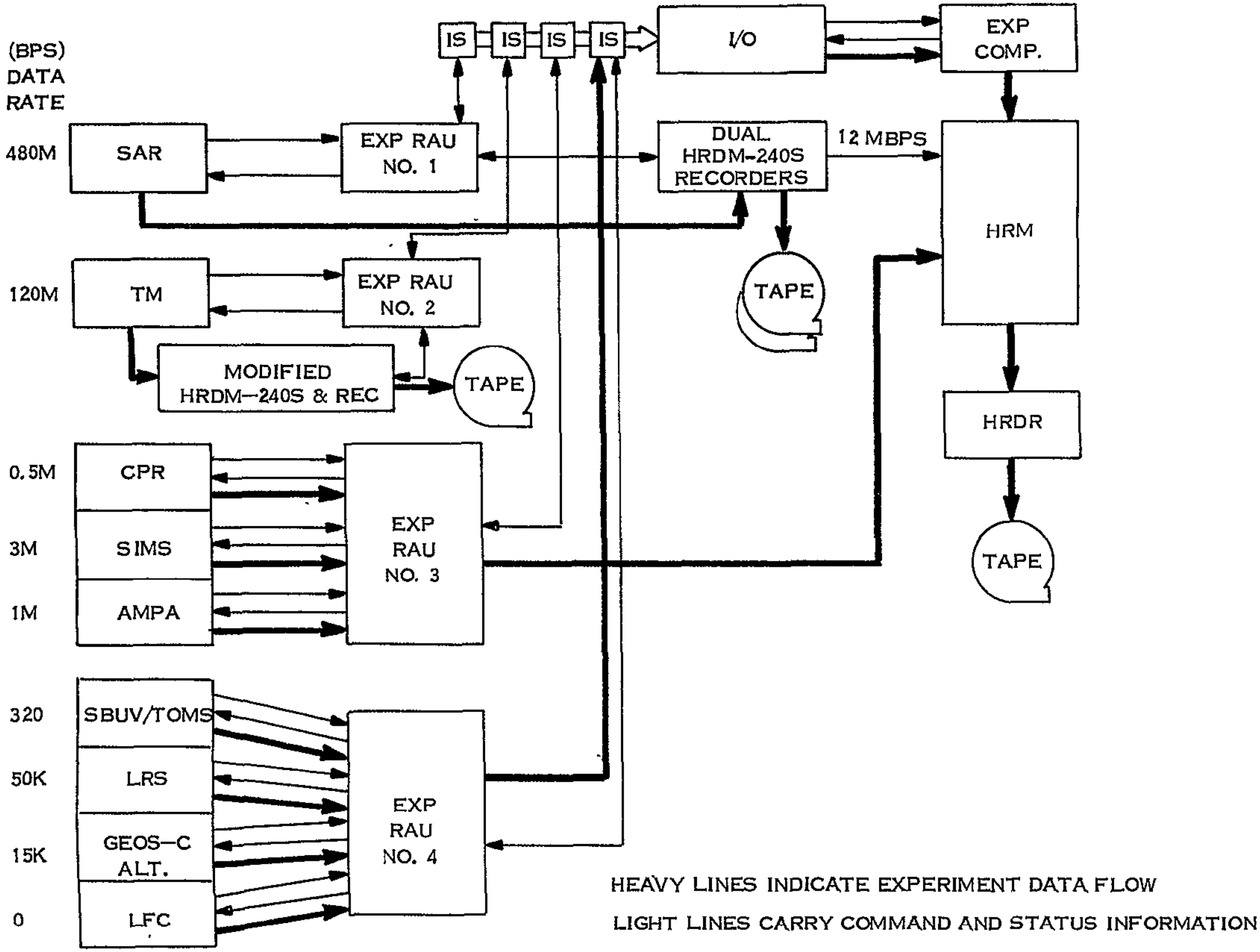


Figure 2-33. Dual Recorder Data Management Solution

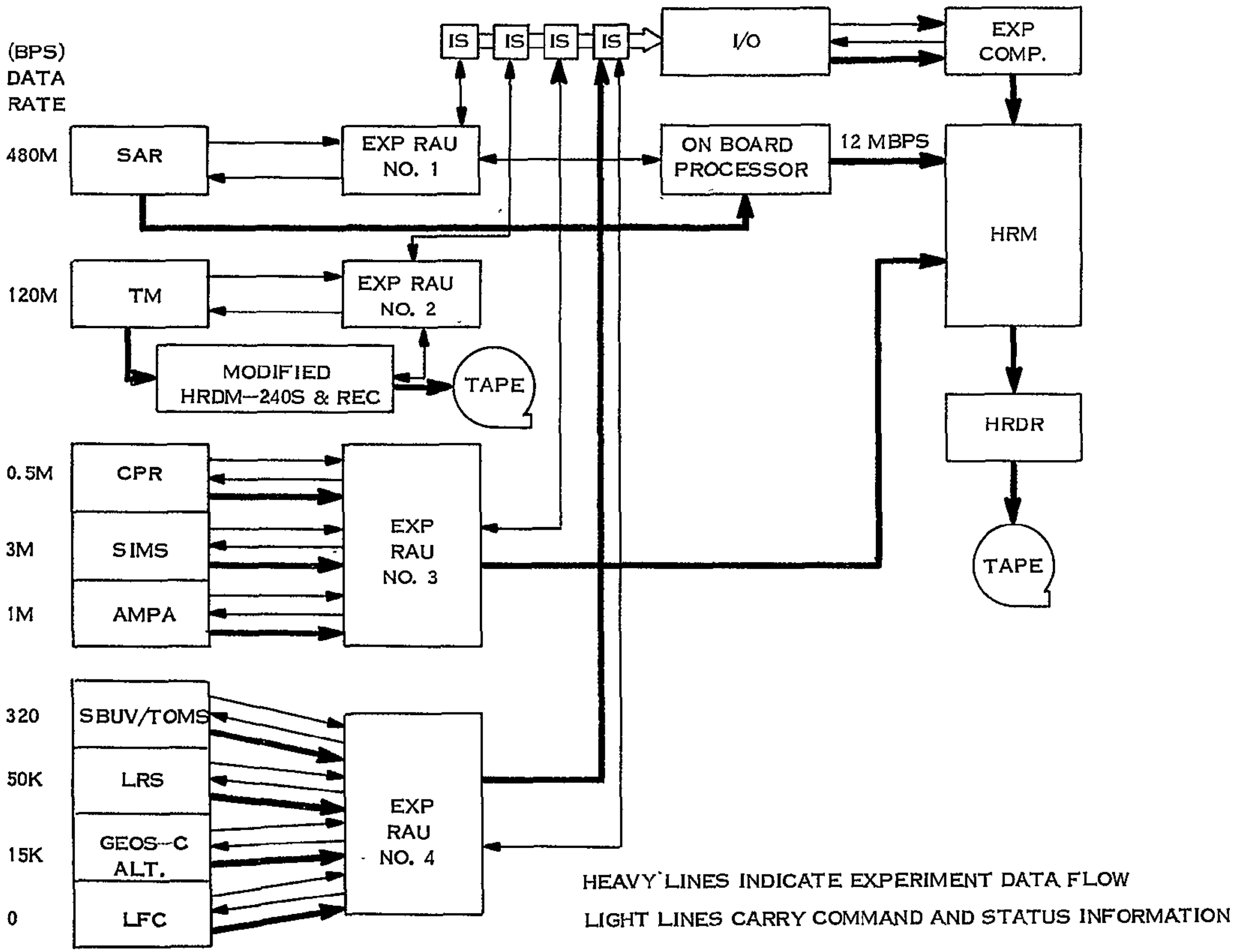


Figure 2-34. On Board Processor Data Management Solution

A slight extension of this second solution is to include processing of TM data (for calibration, and geometric and radiometric correction - but not data compression), and other sensor data (to provide quick look formats and data compression). This possible growth system is shown in Figure 2-35. It should be noted that this extended system is more for convenience than substance, since a recorder is still required for the TM data - which is returned on tape.

2.5.2 POWER ANALYSIS

The power requirement for the payload is a function of the supporting systems (Spacelab and STR), the mission dependent equipment required in the Spacelab for accomplishing the experiments (i.e., racks, cold plates, pointing systems, etc.), and the sensors themselves. The power requirements are obtained from the mission timelines by summing the instantaneous power requirements for the various sensors associated with each experiment throughout their "on" time. This "on" time includes a five-minute warmup, actual operation or data taking, and a three-minute shutdown period. In circumstances where multiple experiments requiring the same sensor are being conducted either simultaneously or in an overlapping mode, double accounting is avoided by considering the power requirement of the sensor once only.

The power requirements for the other elements of the power budget were obtained from reference documents: (1) Spacelab Accommodations Handbook, (2) Space Shuttle System Payload Accommodations, (3) Standard Earth Observation Package for Shuttle.

When all of the above elements are factored into a power profile for this payload, the result is similar to the sample shown in Figure 2-36 for the on-orbit period between 72 and 96 hours. It is observed from Figure 2-36 that there is a steady state level of approximately 5.8 kW required, with peaking to values of 12.4 kW. The steady state level is well within the Shuttle capability of 7 kW average, but the peak values exceed the 12 kW limit quoted for Shuttle payloads. That limit is stated as 12 kW for a maximum of 15 minutes no more often than once in a 3 hour period. The EVAL requirement is for 12.4 kW for 1 to 5 minutes once every 4 to 6 hours. This may well be achievable, although the primary power interface is rated at 12 kW and may not be able to deliver the additional 0.4 kW. If this is the case, peaking batteries will be required.

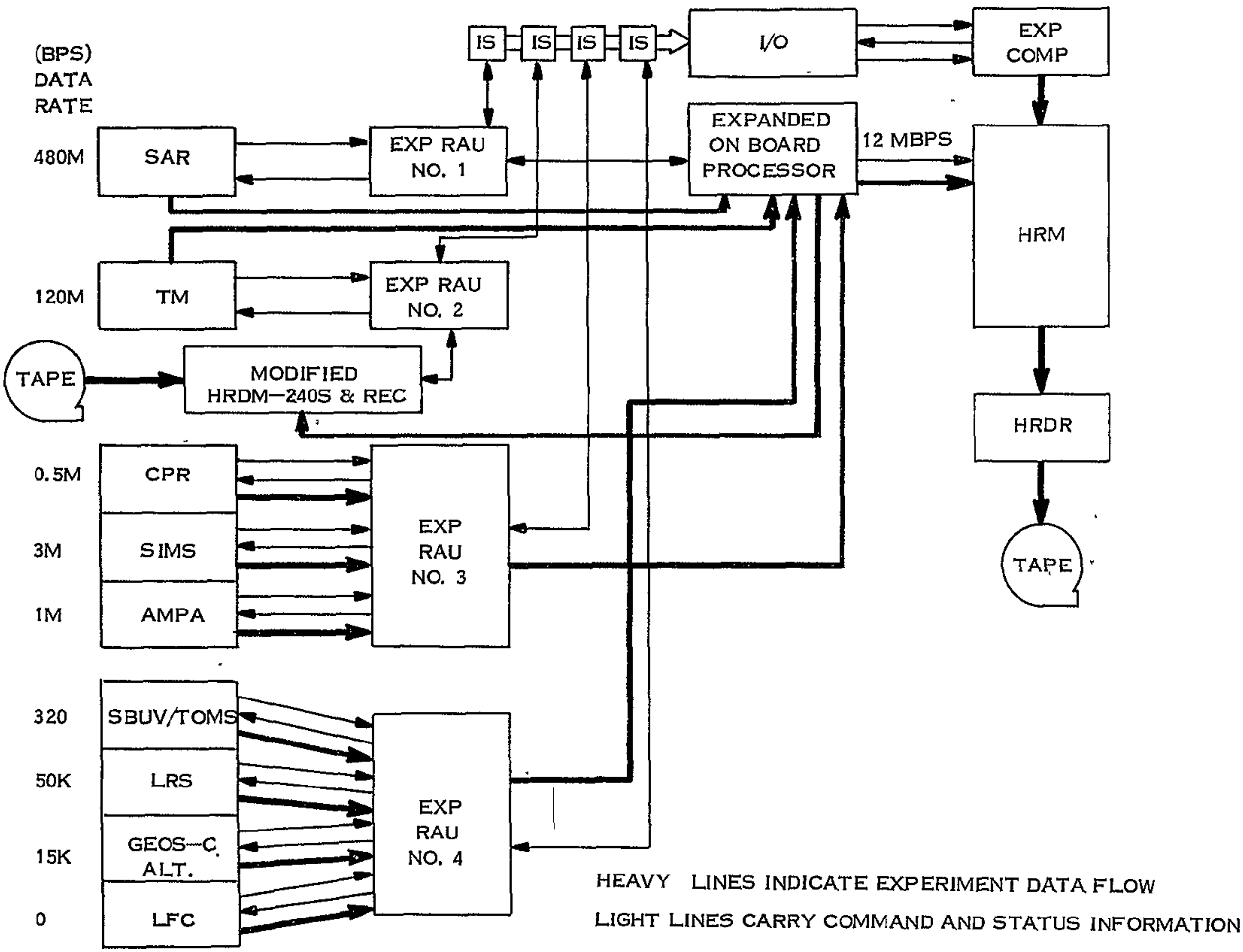


Figure 2-35. Expanded On Board Processor Data Management Solution

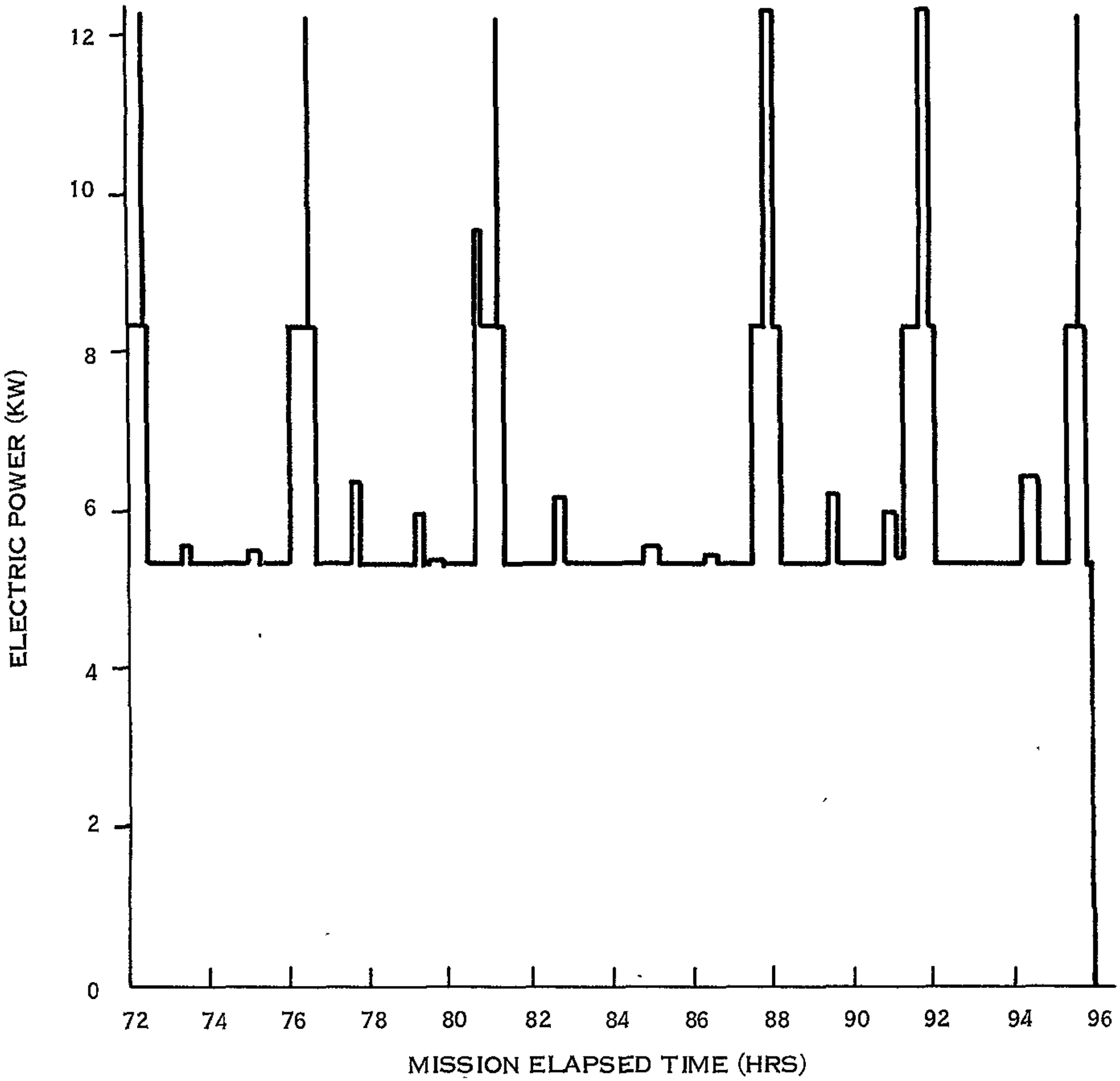


Figure 2-36. Power Profile

A breakdown of the average power and total energy required for each element of the payload is shown in Table 2-17. The dedicated EVAL mission extends to Orbit 119, which is 20 hours more than the nominal 156 hours (7 day) Spacelab mission. During the extended period, the payload is responsible for the electrical energy needs of both Orbiter and payload. Total mission energy requirement is 1252.1 kWh, which is well beyond the 890 kWh provided in the Spacelab baseline. For this reason a second energy kit has been added, raising total capability to 1730 kWh.

Total allowable power/energy (payload and Orbiter) is impacted by radiator heat rejection capability. In the dedicated EVAL configuration the SIR antenna shades a portion of the Orbiter radiator, resulting in degraded average and peak power capabilities. The amount of degradation is as yet undetermined, but it is believed that the 5.8 kW payload average presents no problem. The 12.4 kW peaks are shorter and less frequent than allowable with full radiation capability, and may also be acceptable. Retracting the SIR antenna when it is not in operation may provide the needed heat rejection capability, depending on the thermal capacitance of the system.

Table 2-17. Mission Power/Energy Requirements

Payload Element	Avg Power (W)	Energy (kWh)
Orbiter Aft Flight Deck	168	26.2
STR System	102	15.9
Spacelab Only	3,800	592.8
Mission Dep Equipment	1,151	179.6
Experiment Equipment	598	93.2
Total Baseline	5,819	907.7
Mission Extension (20 hrs)		
Orbiter *	12,000	240.0
Payload **	5,221	104.4
Total Mission	--	1252.1

* Payload chargeable using Orbiter fuel cells

** No experiment operations during mission extension

SECTION 3

STANDARD TEST RACK PAYLOAD

This section addresses the implementation of a candidate Environmental Quality payload using the Standard Test Rack (STR) "Cradle" system as the supporting structure. The payload is considered to be a standby package ready for flight on short notice and a non-interferring, space available basis. For the purposes of this study, Shuttle mission number 22 as described in the Early STS Mission Plan, dated June 22, 1976, prepared by the Marshall Space Flight Center, has been chosen as representative of the type of flight on which such a system could be flown. This particular flight is identified with a launch date of October 1981 and orbital parameters of 460 Km altitude and 33° inclination. The primary objective of mission 22 is to deploy a Very Long Baseline Interferometer (AS-05-A), deploy a Gravity Probe (AP-04-A), and retrieve a Solar Max spacecraft (SO-03-A). Mission duration is planned for seven days.

The proposed Environmental Quality/STR payload would occupy unused space around the retrieved spacecraft. There would be no physical interference with either the deployment or retrieval of the companion payloads because of the STR structure design and positioning. Operational interference would be avoided by scheduling dedicated mission times for deployments, retrieval, and operation of the Environmental Quality package. Essentially no crew involvement is required, and the location of the STR payload will result in only a negligible change in the total center of gravity.

The basic mission of this EVAL/STR Environmental Quality payload is to investigate some of the causes and effects of tropospheric pollution. In particular, information regarding the tropospheric concentration of minor constituents and harmful pollutants is desired on a global scale. Performance of this mission will result in the operational demonstration of the techniques and sensors which will ultimately be flown on a free-flying satellite to continuously monitor tropospheric pollution. Specific objectives of this mission include:

- Determination of background concentrations of the minor constituents and pollutants

- Determination of the variations in the ambient concentrations
- Identification of the sources and sinks of the pollutants
- Determination of the horizontal and vertical transport of the pollutants
- Determination of the radiative interactions between the stratosphere and troposphere

It should be noted that other Environmental Quality payloads, as well as experiment and sensor combinations for most other disciplines, could be developed as STR payloads for mission 22 -- or many other missions. Examples of some additional possible STR payloads are:

1. An Environmental Quality mission directed at stratospheric phenomena. By grouping several small sensors such as HALOE and SER (extinction photometers), LACATE (a scanning spectral radiometer), and HSI (a solar interferometer) on a STR structure, profiles of various constituents can be obtained.
2. A Weather and Climate STR payload could be constructed involving a spectral radiometer for measuring the solar constant and variations in solar radiance, a spectral photometer such as SBUV/TOMS for measuring solar backscattered energy, and active and passive sensors such as a laser ranging system and a scanning imaging radiometer for cloud mapping.
3. Data for a variety of Earth Resources missions could be obtained by combining a scanning spectral radiometer (Thematic Mapper) with a high resolution mapping camera and a panoramic scanning film camera. Such a complement of sensors could easily be accommodated by STR and provide data for missions involving mineral exploration, timber inventories, crop surveys, and urban planning.
4. Communications and Navigation experiments in search and rescue, millimeter wave propagation, and bandwidth compressive modulation could be configured into a set of relatively small antennas (nominally 1m diameter) which would be compatible with the STR dimensions and capabilities.

The advantages of this type of payload is simplified integration and increased flight opportunities. These characteristics are particularly important when the experiment is an evolving one in which multiple flights are desired. In this sense, the Environmental Quality payload considered in this section is a representative candidate.

3.1 STANDARD TEST RACK (STR)

The Standard Test Rack (STR) is a system of standardized modular elements, both structural and functional, designed for mission flexibility and ease of Shuttle integration. It is basically a simple structure for mounting small, automated sensors which can operate in the vacuum of space. It is also selectable in that it consists of an "erector set" of modular components which can be rapidly assembled to meet a variety of mission opportunities -- see Figure 3-1. This versatility results in the possibility of flying on approximately 90% of all Shuttle flights. (One or more of the three configurations shown in Figure 3-1 could be flown on 452 out of 501 Shuttle payloads described in the NASA Traffic Model of January 1974 and the Integrated Mission Plan for the First Two Years of Shuttle Missions.)

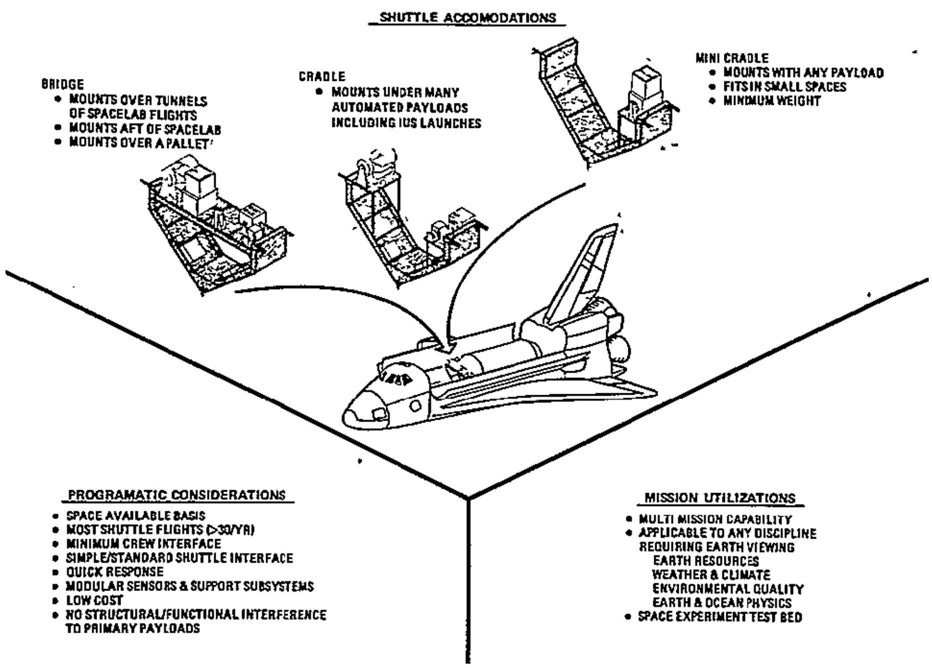


Figure 3-1. STR Configurations

The structure incorporates ancillary sensors for position determination and subsystems for power distribution, thermal control, and data handling. The STR is also capable of accommodating small pointing and stability systems such as MSFC's Miniaturized Pointing Mount (Minimount) for experiments requiring pointing and/or stability in excess of the Shuttle's inherent capabilities of ± 0.5 degrees (can degrade to as high as ± 2.0 degrees dependent upon location in the bay) and ± 0.1 degrees/axis. Single, standardized electrical and mechanical interfaces with Shuttle are also provided. These features result in a highly autonomous payload which is easily integrated and has minimum impact on any other payload it accompanies.

3.1.1 STR SUPPORT SUBSYSTEMS

The STR support subsystems provide attitude and structural support, control the temperature, furnish conditioned electrical power, and manage and process the flow of information to and from the sensors. The support subsystems include:

1. Attitude Subsystem. Consists of:

   a. Gyro Package. Provides angular rate measurement of 0.0001 degrees/sec.

   b. Star Tracker. Provides STR attitude update to <0.5 degrees for pitch, roll and yaw.

   c. Processor. Used for STR attitude determination (the processor is part of the data management and processing subsystem).

2. Power Subsystem. Regulates and distributes the Shuttle main DC-2 bus power.

3. Data Management And Processing Subsystem (DM&PS). Includes all signal management functions and is the controller and executor of all command and output functions for the entire STR system. Specific functions performed by the DM&PS are:

   a. Sensor and subsystem command generation

   b. Housekeeping data formatting and processing

   c. System checkout and operational evaluation

   d. Sensor data processing

   e. Recording and transmission control

   f. Signal routing and path establishment

   The baseline STR also provides the capability for accommodating two types of tape recorders for data storage:

   - Wideband high data rate (240 Mbps) tape recorder

   - NASA standard (10^8 - 10^9) low rate, narrowband tape recorder

4. Thermal Subsystem. Controls component surface temperature to a maximum average of 21°C and minimum of 5°C using a passive and louver system.

5. Structure Subsystem. This subsystem consists of a Strongback which provides the base for the Bridge, Cradle, and Mini-Cradle. It allows for the clearance around the Spacelab Tunnel. The Strongback is U-shaped, and transmits the STR loads to the trunnion fittings. The various modular configurations are shown in Figure 3-2, while their capabilities are summarized in Table 3-1.

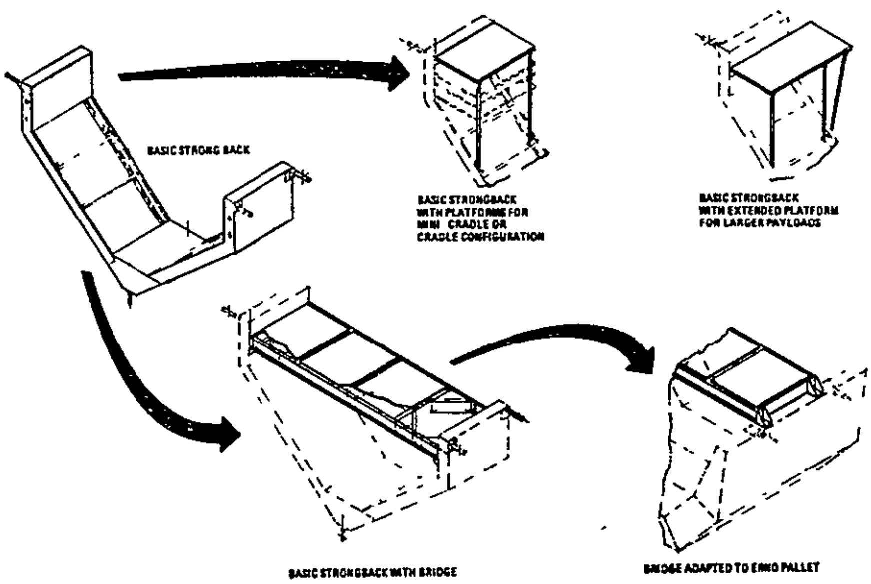


Figure 3-2. Modular STR Structure

Table 3-1. Summary of STR Structures

Configuration	Volume Availability (m^3)	Mounting Surface (m^2)	Payload Capability (Kg)	Structure Weight (kg)
BRIDGE 3937 x 1219 mm	23.7	16	1043	313
CRADLE 1676 x 1219 mm	6.2	7.0	1108	252
MINI CRADLE 838 x 1219 mm	4.4 (max)	6.0 (max)	676	232

3.1.2 STR INTEGRATION

One of the primary advantages of the STR is that it can be physically integrated into Shuttle after all other payloads are in place - 24 hours prior to launch. This is achieved by integrating and testing all of the STR modules in the STR ground facility located near the launch pad. This activity, along with software development and mission analyses involving operating timelines, resultant center of gravity, and resource allocations for the STR operating in conjunction with the primary payload, is accomplished over a period of six weeks prior to flight during Level I and II integration. The STR is then transported to the pad and:

1. The electrical connectors between each section are disconnected.
2. The structure is mechanically disassembled into three sections (bridge configuration) or two sections (cradle configuration).
3. The structure is installed around a primary payload and reassembled to its original configuration (see Figure 3-3).
4. All electrical connectors are reconnected.
5. Power is applied to the STR and a short operational check is performed.

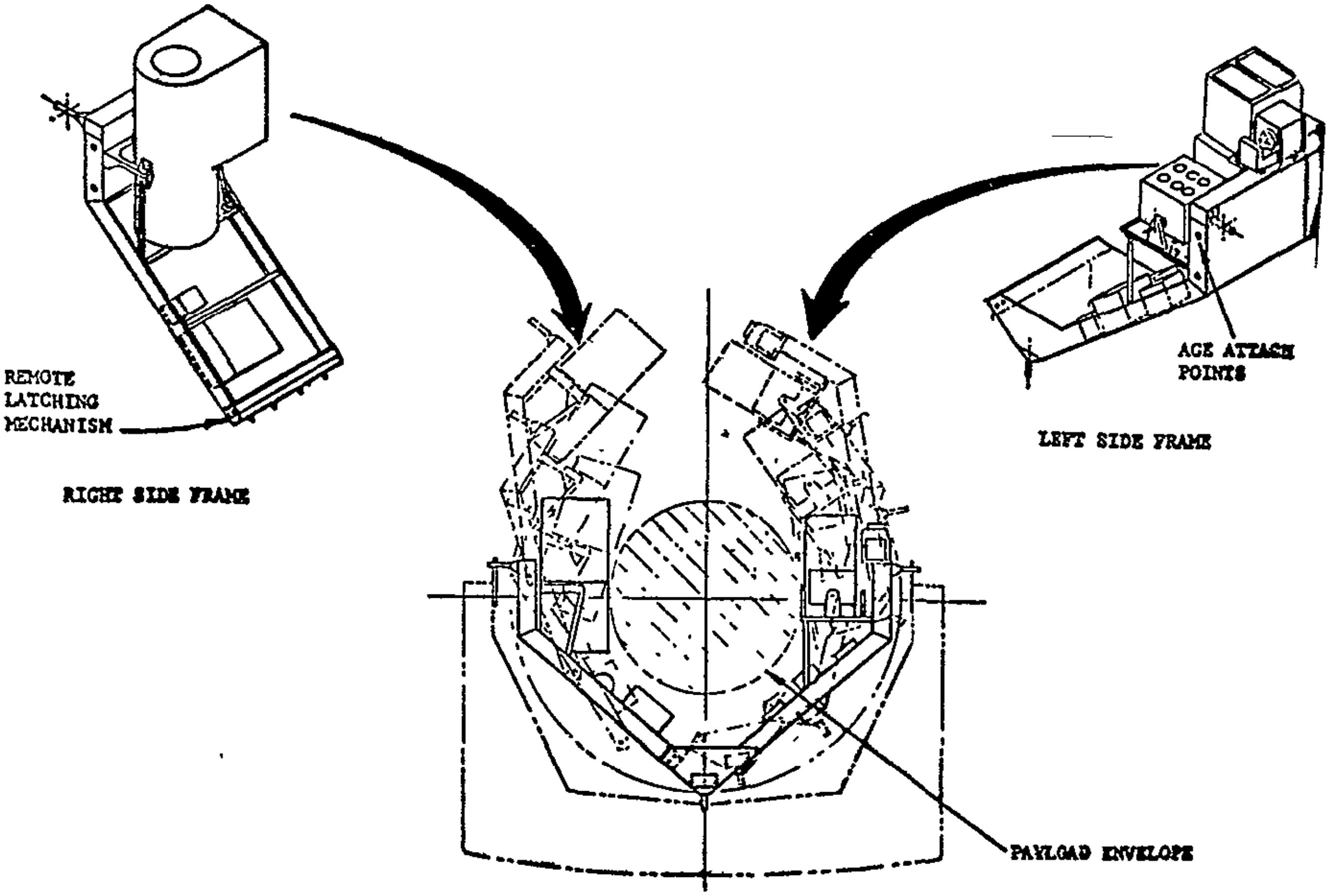


Figure 3-3. Installation of the STR Cradle

3.1.3 STR OPERATION

The functional flow for STR ground operation (depicted in Figure 3-4) is:

1. Select a STR mission(s) for a specific Shuttle flight.
2. Correlate the available weight, volume, and location with the current file of user data requests.
3. Assemble, calibrate and test the selected hardware at the STR ground facility.
4. Generate necessary software for orbital operation.
5. Install STR into the Shuttle and perform a simple operational check.
6. On-orbit operation with non-interference with primary payload and minimum crew participation.
7. Post landing, remove STR prior to removal of primary payloads.
8. Return hardware to STR facility for refurbishment, recalibration and preparation for the next flight.
9. Check quality of data at the STR quick-look facility (first step in processing); send data to a user data processing facility for final processing and distribution to the users.

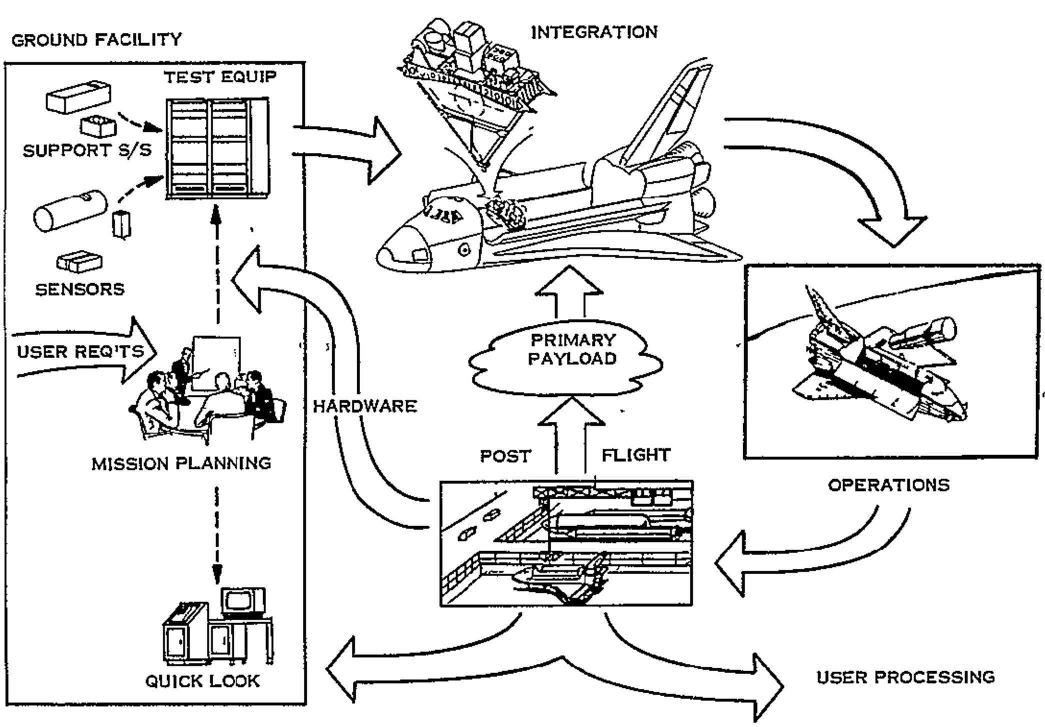


Figure 3-4. STR Ground Operations

3.2 PAYLOAD DESCRIPTION

Sensors associated with the tropospheric pollution mission include Monitoring of Air Pollution from Satellites (MAPS), Correlation Interferometry for Measurement of Atmospheric Trace Species (CIMATS), Solar Backscatter UV and Total Ozone Mapping Spectrometer (SBUV/TOMS), and the High Resolution IR Spectrometer (HIRS). The location of these sensors on the STR structure is shown in Figure 3-5. Scientific designation and engineering details describing the four sensors are provided in Tables 3-2 and 3-3, and the following paragraphs.

3.2.1 MONITORING OF AIR POLLUTION FROM SATELLITE (MAPS)

This instrument measures concentrations of CO, CO_2, SO_2, NO, NO_2, NH_3, and CH_4 in the range of 0.001 ppm to 350 ppm. Optical correlation of gases through a gas filter correlation analyzer permits selective measurement of the change in infrared radiation in the 2 to 20 micron range due to specific pollutants. The measurement of tropospheric pollutants will permit the determination of constituent dispersal rates and longterm buildup to forecast regional pollution and establish relations with global meterology. Chemical processes and sink mechanisms in the upper atmosphere will also be investigated through MAPS measurements. Figure 3-6 indicates the MAPS configuration.

3.2.2 CORRELATION INTERFEROMETRY FOR MEASUREMENT OF ATMOSPHERIC TRACE SPECIES (CIMATS)

The instrument is a two channel interferometer, one operating in the non-thermal infrared (2 to 2.4 μm) and the other in the thermal infrared (4 to 9 μm). A PbS detector operating at 195^{o}K is used in the 2 to 2.4 μm channel and a HgCdTe detector operating at 77^{o}K in the 4 to 9 μm channel. Each channel is capable of containing five (5) narrowband filters, thus providing the capability of making ten different spectral measurements. Two measurements, one in each channel, are made simultaneously with a measurement time of one second. This sensor has the capability of operating both in the nadir viewing mode and in the earth limb viewing mode.

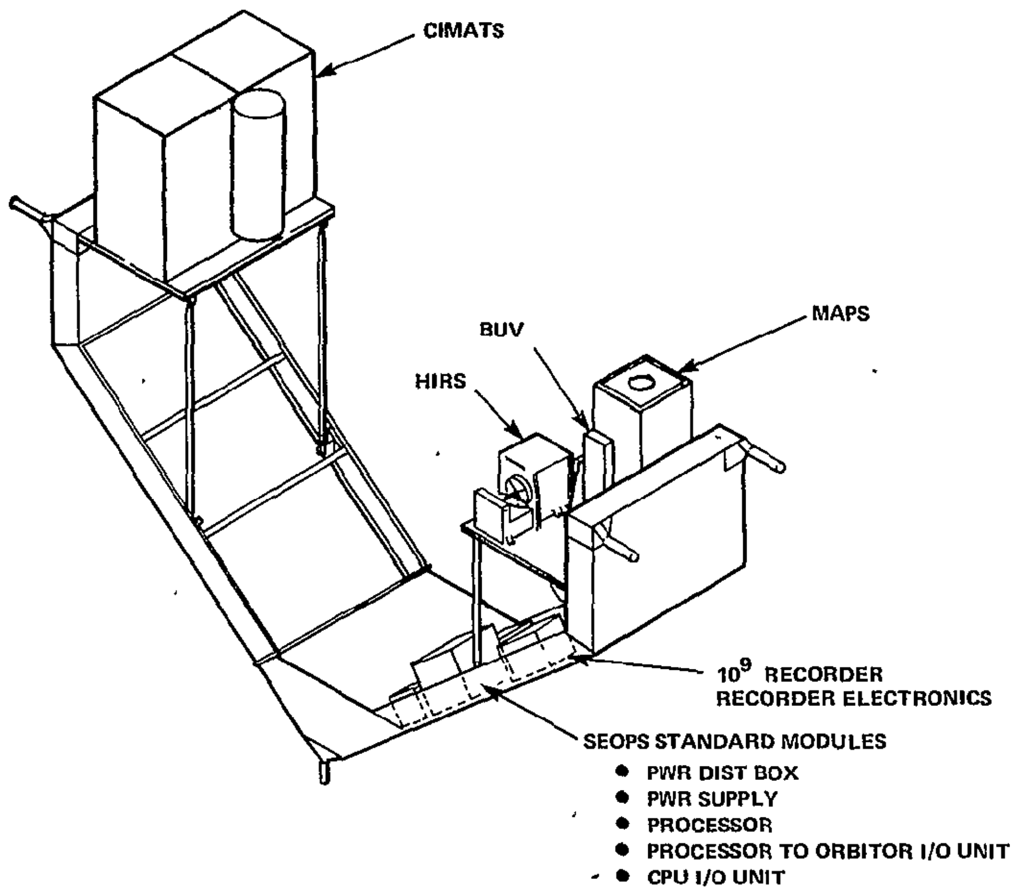


Figure 3-5. STR Environmental Quality Cradle Configuration

Table 3-2. Payload Description

	Type	Objective	Spectral Bands No.	Spectral Bands Location	Viewing
Correlation Interferometer for Measurement of Atmospheric Trace Species (CIMATS)	Correlation Interferometer	Measure CO, CH_4, N_2O, NH_3, O_3, SO_2 H_2O	5 5	2-2.4 um 4-9 um	Nadir Viewing
Measurement of Air Pollution from Satellites (MAPS)	Gas Filter Radiometer	Measure global distribution of CO, SO_2, CH_4, NH_3	3	3-10 um	Nadir Viewing
Backscatter Ultraviolet Spectrometer (BUV)	Grating Spectrometer	Measure back scattered solar ultraviolet radiation	12 1	0.25-0.34 um 0.38 um	Nadir Viewing
High Resolution IR Radiometer (HIRS)	Scanning Spectral Radiometer	Measurement vertical temperature profile and H_2O distribution	17	0.7-15 um	Nadir Viewing

Table 3-3. Payload Support Requirements

	Size, CM			Weight	Power (W)		Data Rate	Field of View	View Angle	Pointing Accuracy	Stability Amplitude
	L	W	H	(Kg)	AVE	PEAK	(BPS)			(DEG)	(ARC SEC)
Correlation Interferometer for Measurement of Atmospheric Trace Species (CIMATS)	(Sensor) 60 (Telescope) 18 (Electronics) 50	 35 18 50	 38 36 20	50	180	185	2916	7°	7°	5 (2 knowledge)	36
Measurement of Air Pollution from Satellites (MAPS)	37 (Electronics) 32	37 32	50 20	41	67	-	840	7°	7°	5 (2 knowledge)	360
Backscatter Ultraviolet Spectrometer (BUV)	20 15	15 15	55 15	16	20	-	220	12°	12°	2.0	100
High Resolution IR Radiometer (HIRS)	52	26	43	45	20	-	3389	1.5°	72°	5 (2 knowledge)	360

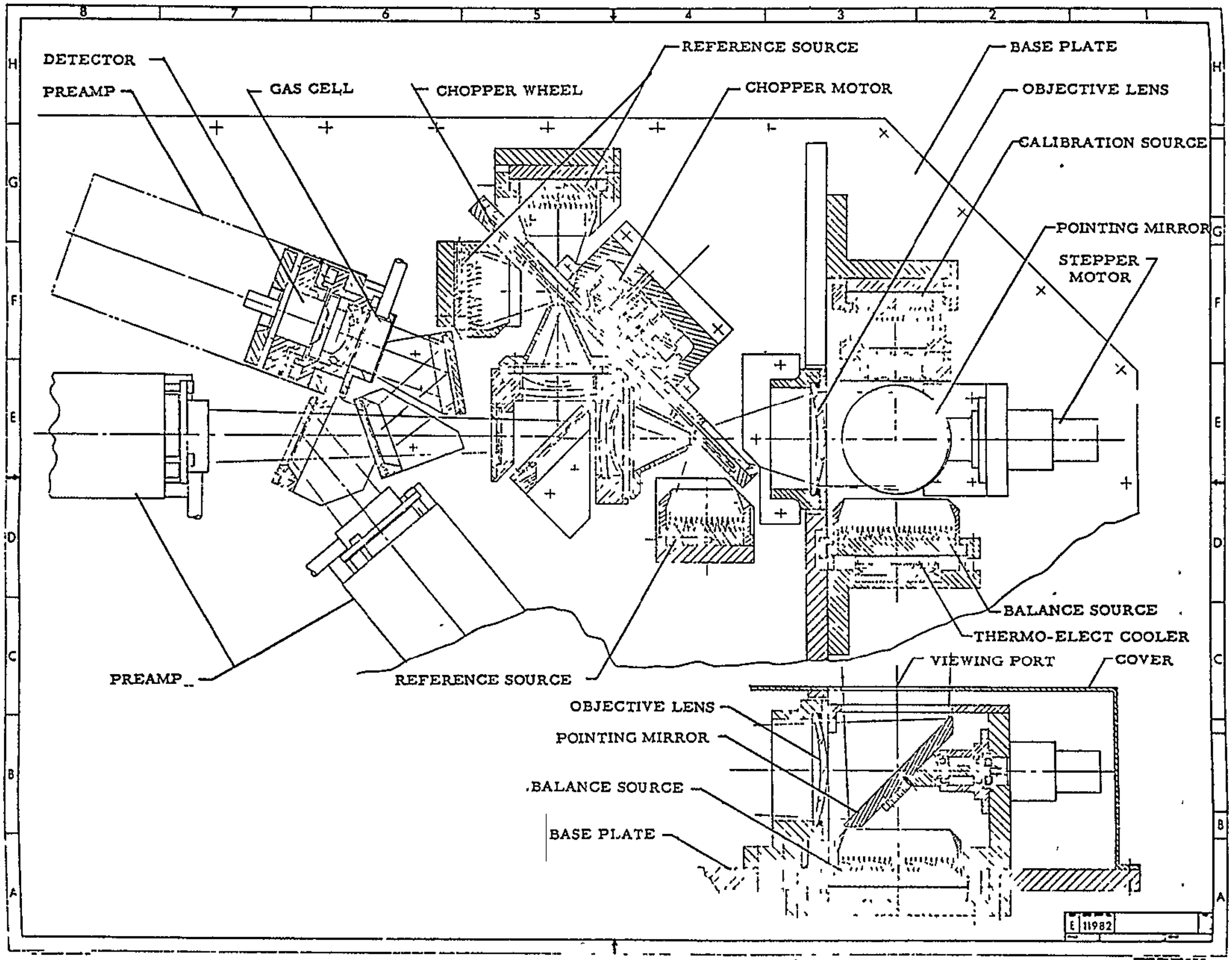


Figure 3-6. MAPS Configuration

Figure 3-7 is a schematic of the CIMATS sensor which is presently undergoing laboratory testing. Two interchangeable telescopes are available, a 7° unit weighs 5 kg and is 40 cm long x 20 cm in diameter and a 2° unit weighing 14 kg and 56 cm long x 36 cm in diameter. The electronics operate from 28 volts D.C., weigh 9 kg and are contained in a box 18 x 23 x 30 cm. Analog to digital conversion of the experimental data is accomplished in the electronics.

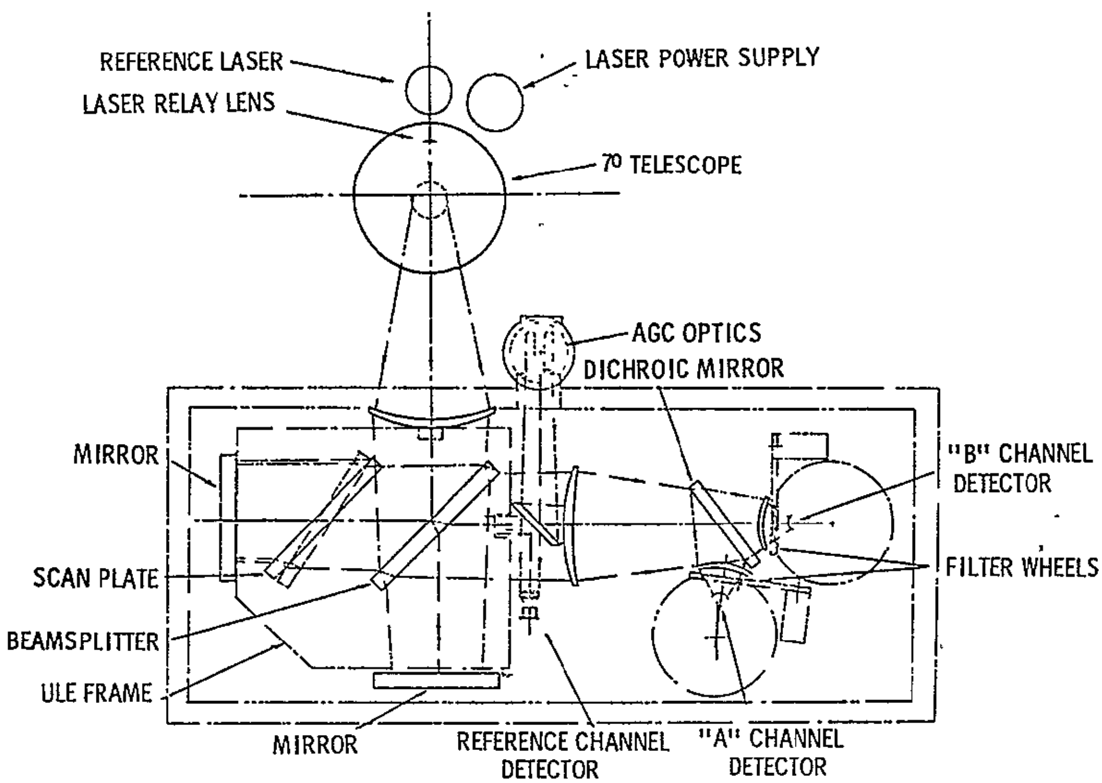


Figure 3-7. CIMATS Sensor Schematic

3.2.3 SOLAR BACKSCATTER UV & TOTAL OZONE MAPPING SPECTROMETER (SBUV/TOMS)

The SBUV/TOMS provides both total synoptic and sampled vertical ozone distributions to an altitude of 60 Km. The UV spectrometer measures solar UV that is back-scattered by the earth's atmosphere at 12 wavelengths between 2500°A and 3400°A with a spectral bandpass of 10°A.

The ozone mapper, operated in parallel with the UV spectrometer, has a step scan ± 0.89 rad, normal to the orbital track with an IFOV of 0.052 rad. At each scan position the earth's radiance is monitored at four wavelengths between 3100 and 3400 A°, and at 3800 A° to infer the total ozone amount. The sensor is configured as shown in Figure 3-8.

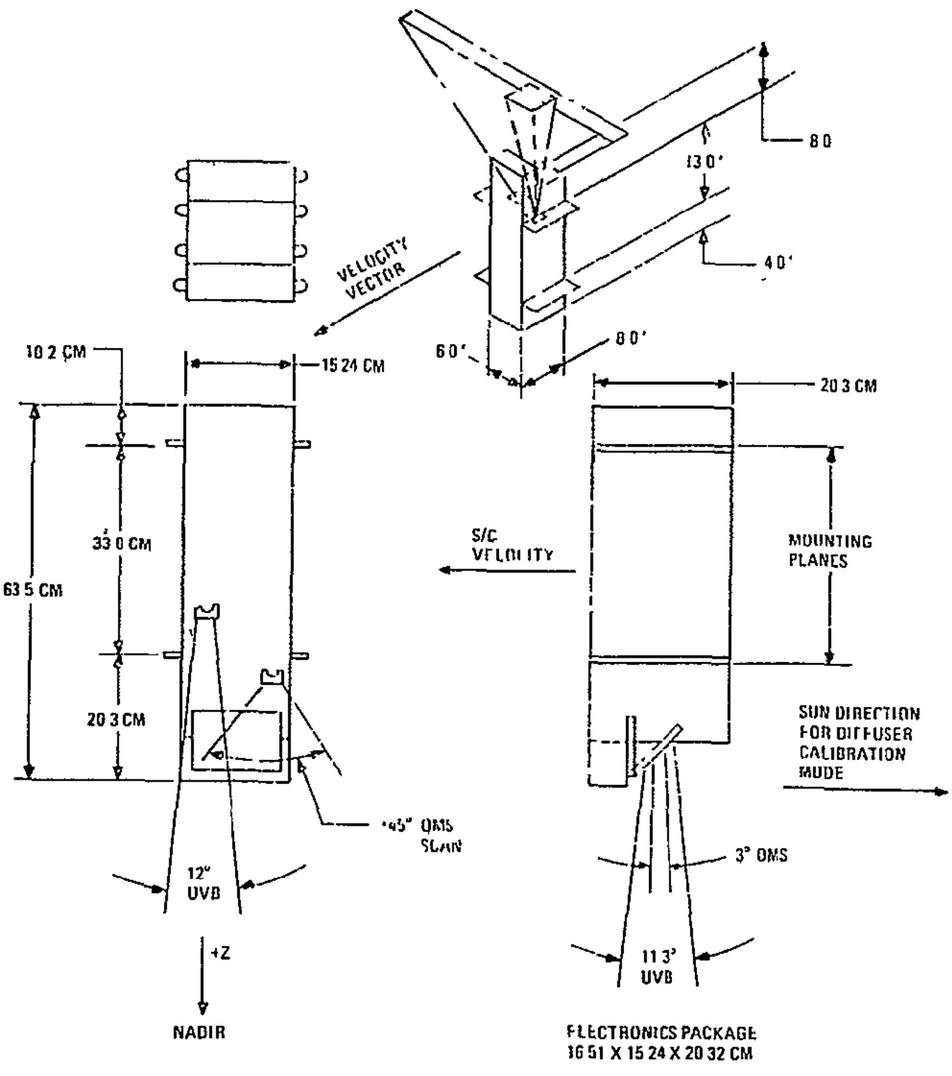


Figure 3-8. SBUV/TOMS

3.2.4 HIGH RESOLUTION IR SPECTROMETER (HIRS)

The Infrared Spectrometer is designed to obtain spatially independent IR radiances (that are unbiased with respect to cloud condition) at sufficient spectral and spatial resolutions so that the data may be used for determining the thermal structure of the earth's atmosphere. This instrument is a modification of the sensor currently in operation on Nimbus-F.

Basically, HIRS is a filter wheel device which scans normal to the orbit plane with a scan angle of $\pm 36.9^{\circ}$ about the nadir for earth view. The optical telescope focuses the received radiant energy onto two cooled detectors and a photodiode which is used as a visible energy channel. Prior to reaching the detectors, the energy is spectrally separated into long wave (LW), short wave (SW), and a visible component, chopped and bandpass filtered. There are three detectors and 17 spectral bandpass filters. Figure 3-9 provides a sketch of this sensor.

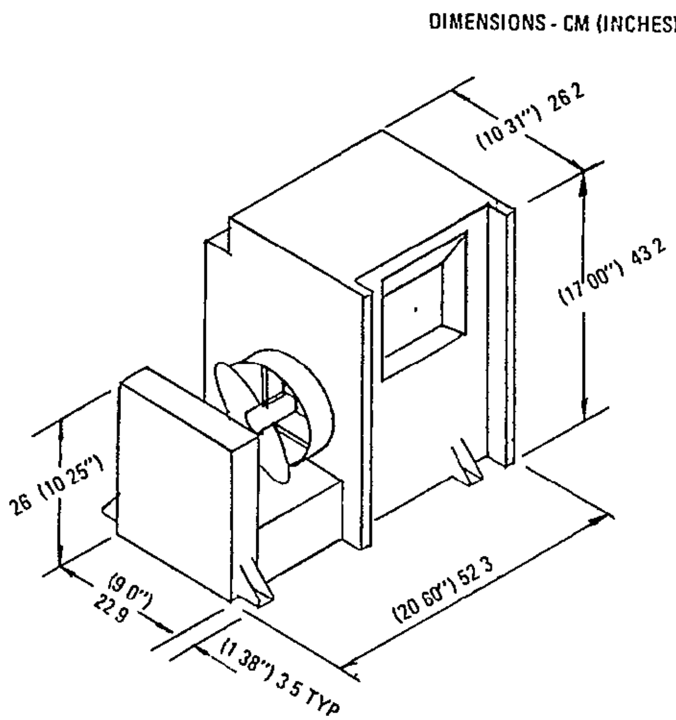


Figure 3-9. High Resolution Infrared Spectrometer

The HIRS sensor provides temperature and water vapor profiles up to 40 km. These data are essential in analyses and interpretation of data from the CIMATS and MAPS sensors as well as being valuable for meteorology applications.

3.3 PHYSICAL ACCOMMODATIONS

The physical accommodation of payload equipment on the STR cradle is a straight-forward design. Payload mounting area is limited but is ample for the four instruments of the EVAL/STR payload. All instruments view along the nadir and present no field of view conflicts. Weight and balance constraints are met through proper placement of the STR in the payload bay. The EVAL/STR payload meets all design constraints and is a good example of the minimum interface, space available utilization that is the hallmark of the STR concept.

3.3.1 PAYLOAD WEIGHTS AND LOCATIONS

Payload and payload chargeable weights of experiments, STR, Payload Specialist Station (PSS), and contingency allowance are summarized in Table 3-4, along with weights of the primary delivery/retrieval cargo. The total cargo launch weight of 20,790 kg (including 503 kg for the EVAL/STR payload) is well within the Shuttle capability for the desired orbit if an Orbit Maneuvering System (OMS) kit is added in the aft end of the cargo bay (see Figure 3-10). Note that the OMS kit is required for the baseline mission even without the addition of the EVAL/STR payload.

In the reference mission analysis for the baseline mission, provision was not made for the OMS kit. This kit weighs about 6000 kg and protrudes some 3m into the payload bay. Hence, the addition of the OMS kit makes launch weight marginal and precludes Gravity Probe B installation in the location shown. Since the intent of this study is to evaluate the addition of EVAL/STR to an existing mission and not to redesign the existing mission, the OMS kit requirement is set aside and the baseline requirement is assumed valid as given.

A payload layout drawing is shown in Figure 3-11. Significant features of the drawing are:

1. The STR cradle is installed between the two support frames used for SMM spacecraft retrieval.
2. The CIMATS, MAPS, BUV, and HIRS are all hard mounted to the STR and do not independently point or slew.
3. The EVAL/STR provides adequate clearance for SMM retrieval and stowage.

Table 3-4. Payload and Payload Chargeable Weights

	Launch Weight (kg)	Landed Weight (kg)
Experiment Sensors	152	152
CIMATS	(50)	(50)
MAPS	(41)	(41)
BUV	(16)	(16)
HIRS	(45)	(45)
STR	302	302
Structure	(252)	(252)
Subsystems	(50)	(50)
Other Payload Chargeable Weights	49	49
Payload Specialist Station (PSS)	(25)	(25)
Payload Weight Contingency	(24)	(24)
Total EVAL/STR Payload	503	503
Very Long Baseline Interferometer	16473	1657
First Stage IUS	(10395)	(0)
Second Stage IUS	(3449)	(0)
Shuttle Interface	(1657)	(1657)
Payload Adapter	(64)	(0)
VLBI Spacecraft	(907)	(0)
Gravity Probe B	1425	775
Pallet	(600)	(600)
Platform	(95)	(95)
Attach Structure	(80)	(80)
GPB Spacecraft	(600)	(0)
Solar Maximum Mission	959	2549
Attach Structure	(864)	(864)
Thermal Control & Power Dist.	(95)	(95)
SMM Spacecraft	(0)	(1590)
Other Payload Chargeable Items	1430	737
Retention Fittings	(420)	(420)
Payload Specialist Station (PSS)	(45)	(45)
Payload Weight Contingency	(965)	(272)
Total Primary Cargo Weight	20287	5718
Total Cargo Weight at Launch	20790	
Total Cargo Weight at Landing		6221
Payload Weight Margin at Launch	4210*	
Payload Weight Margin at Landing		8294**

* Based on 25,000 kg launch weight capability

** Based on 14,515 kg landing weight limit

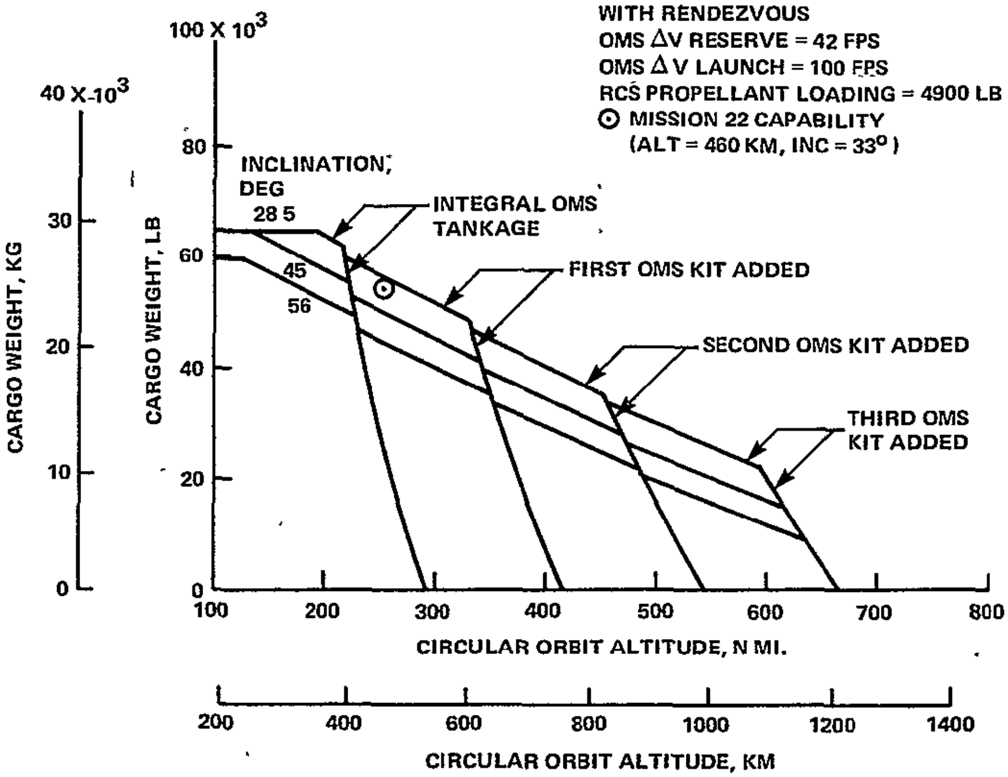


Figure 3-10. Cargo Weight vs. Circular Orbital Altitude - KSC Launch, Delivery and Rendezvous

3.3.2 PAYLOAD CENTER OF GRAVITY

The aerodynamic flight phases of the Shuttle Orbiter (entry and landing, boost phase abort) place rigid center of gravity constraints on Shuttle payloads. The most severe are the X-axis limits which require payload cg to be in the aft portion of the payload bay, and the Y-axis limits which require payload cg to be within a few inches of the payload bay centerline. Z-axis limits are less stringent, allowing cg locations up to 4 feet above or below the payload bay centerline.

All payload chargeable items are included in cg determination, including payload equipment on the Orbiter aft flight deck (at the PSS). Payload cg locations for the EVAL/STR mission are shown in Table 3-5. The payload cg for launch falls far outside the X-axis limit (see Figure 3-12). This is entirely due to the primary delivery cargo. As stated in the previous section, the intent of this study is not to redesign the baseline mission to meet realistic Shuttle

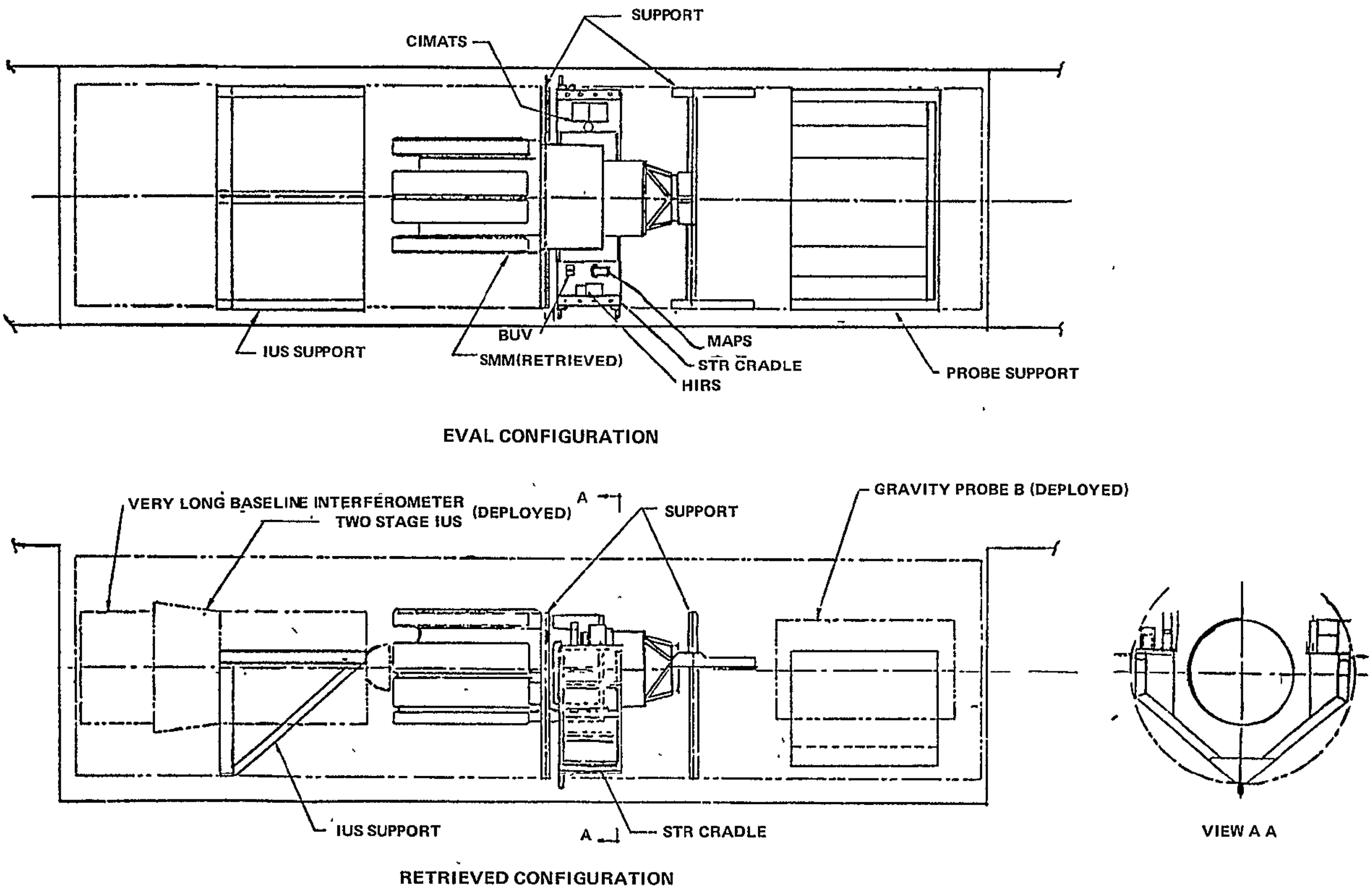


Figure 3-11. Payload Configuration

Table 3-5. Payload Center of Gravity

	Weight (kg)	X_{cg} (m)	Y_{cg} (m)	Z_{cg} (m)
Experiment Sensors				
CIMATS	50	10.30	-1.69	0.82
MAPS	41	10.60	1.50	0.65
BUV	16	9.95	1.48	0.68
HIRS	45	10.30	1.93	0.70
STR				
Structure	252	10.30	0	-0.75
Subsystems	52	10.30	0	-1.25
Other P/L Weights				
PSS	25	- 0.81	0	1.50
P/L Contingency	24	9.78	0	0
Primary Cargo				
VLBI S/C & Adapter (launch)	971	0.95	0	0
Two Stage IUS (launch)	13844	4.75	0	0
Shuttle/IUS I/F	1657	4.50	0	0
GPB S/C (launch)	650	16.10	0	0
GPB Pallet & Structure	775	16.10	0	-1.35
SMM S/C (land)	1590	9.50	0	0
SMM Structure & Systems	959	11.80	0	-0.35
Retention Fittings	420	9.15	0	-0.75
PSS	45	- 0.81	0	1.50
P/L Contingency (launch)	965	5.72	0	0
P/L Contingency (land)	272	9.25	0	0
Center of Gravity at Launch		5.81	0.004	-0.083
Center of Gravity at Landing		9.23	0.014	-0.278

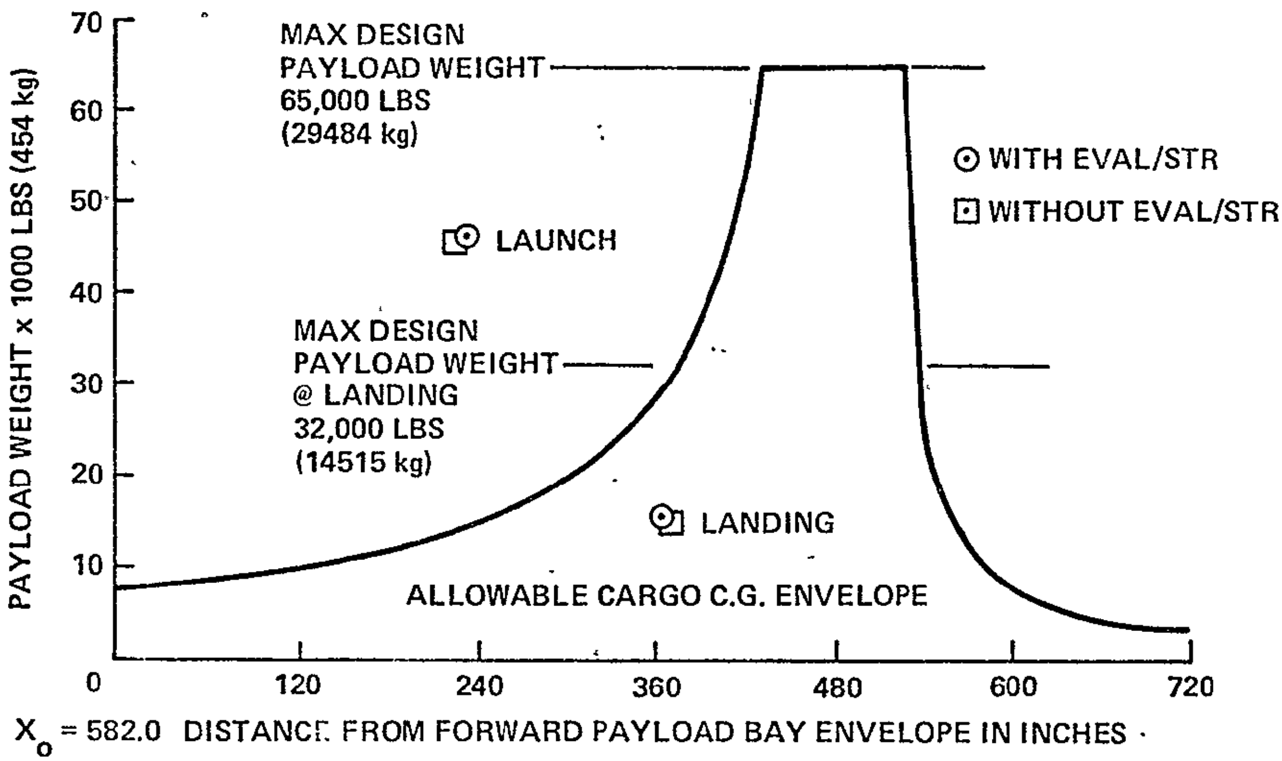


Figure 3-12. Payload CG Along X-Axis

requirements, but to assess the impacts of EVAL/STR on an existing mission. EVAL/STR actually improves payload cg at launch, albeit not nearly enough to bring it within the required envelope. The payload cg at landing is well within X-axis limit.

Payload cg locations in the Y and Z axis directions are well within limits (Figures 3-13 and 3-14). Lateral cg offsets are due entirely to assymetric experiment loading on STR (102 kg on the port side, 50 kg on the starboard) and are insignificant.

3.3.3 FIELD OF VIEW

All EVAL/STR experiments are hard mounted to the STR cradle in such a way that they point along the nadir when the Orbiter flies in the Z-LV attitude. No off-nadir pointing or slewing is required. The EVAL/STR arrangement satisfies all experiment viewing requirements as summarized in Table 3-6.

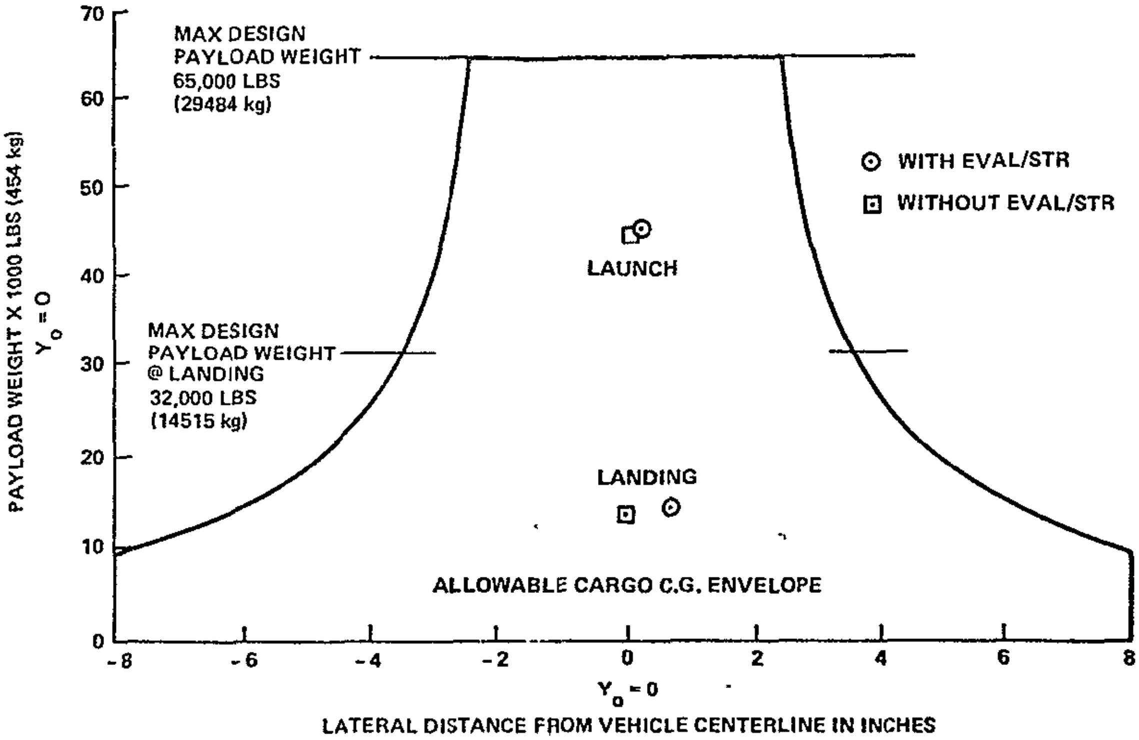


Figure 3-13. Payload CG Along Y-Axis

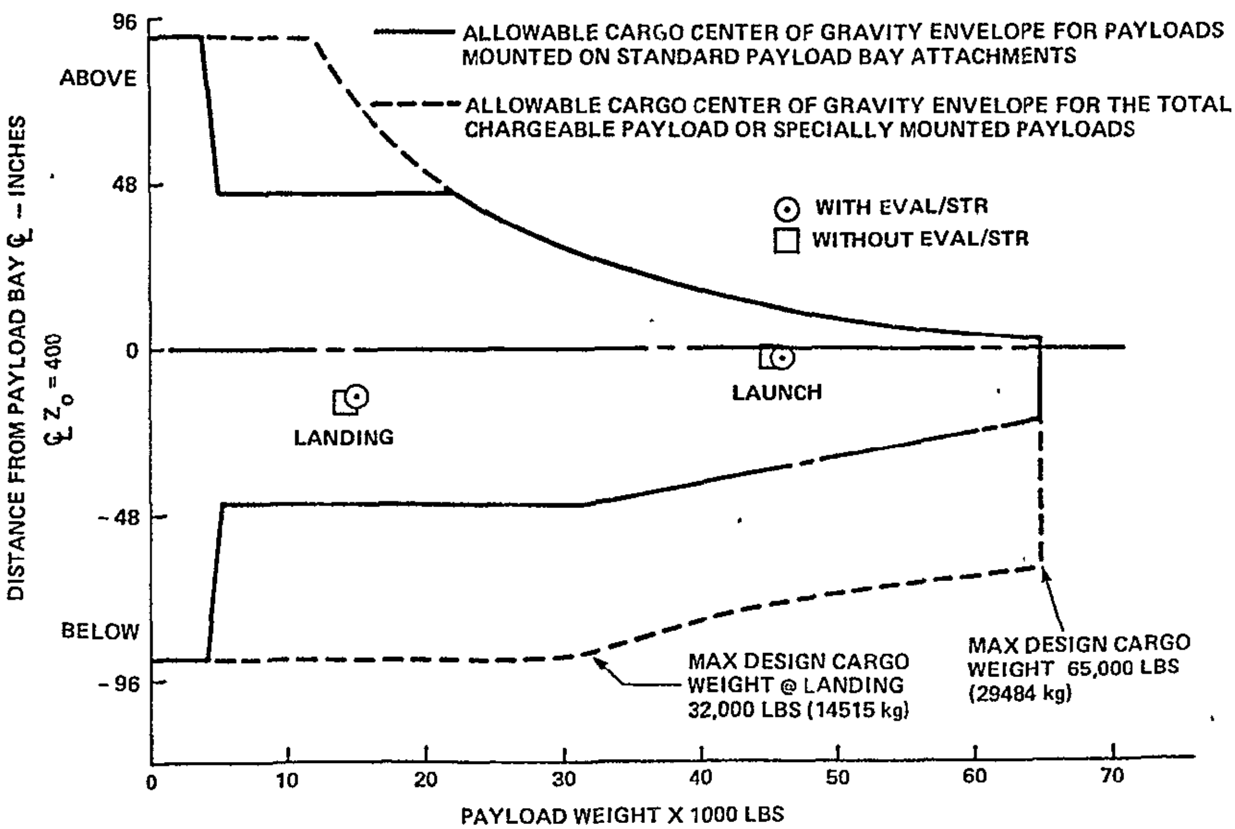


Figure 3-14. Payload CG Along Z-Axis

Table 3-6. Experiment Field of View Summary

Sensor	Viewing Requirement	Viewing Capability
CIMATS	Nadir Viewing 7° Instantaneous FOV 7° Total FOV	Location on starboard side of STR provides unobstructed nadir view
MAPS	Nadir Viewing 7° Instantaneous FOV 7° Total FOV	Location on port side of STR provides unobstructed nadir view
BUV	Nadir Viewing 12° Instantaneous FOV 12° Total FOV	Location on port side of STR provides unobstructed nadir view
HIRS	Nadir Viewing 1.5° Instantaneous FOV 72° Total FOV	Location on port side of STR provides unobstructed nadir viewing

3.3.4 INTERFACES

Payload to Shuttle interface definition is begun with schematic diagrams that define the payload accommodation resources utilized by each experiment. Experiment schematics for CIMATS, MAPS, BUV, and HIRS are given in Figures 3-15 through 3-18. These figures show required connections between experiment and STR subsystems: Electric power, command/telemetry, data, C & W, and mounting connections are defined.

Once the individual experiment interfaces have been identified, the combined payload to STR/Shuttle interfaces can be investigated. This is accomplished with a system schematic as shown in Figure 3-19. Payload equipment is assigned to a specific location and all required connections are shown. Each connection is analyzed to ensure that combined payload requirements are compatible with the payload accommodations capabilities of the carrier element (STR). When compatibility is ensured, detailed interface design can proceed.

The EVAL/STR payload shows no interface incompatibilities. Electric power, command/telemetry, and caution and warning connections between STR and Orbiter are provided through utility service panels on the forward bulkhead of the cargo bay (Sta 576) and on the starboard side wall (Sta 695). These stations provide for multiple access and are shared with the primary cargo. No conflicts are seen since Shuttle resources are adequate for all cargo requirements even if concurrent operations are considered.

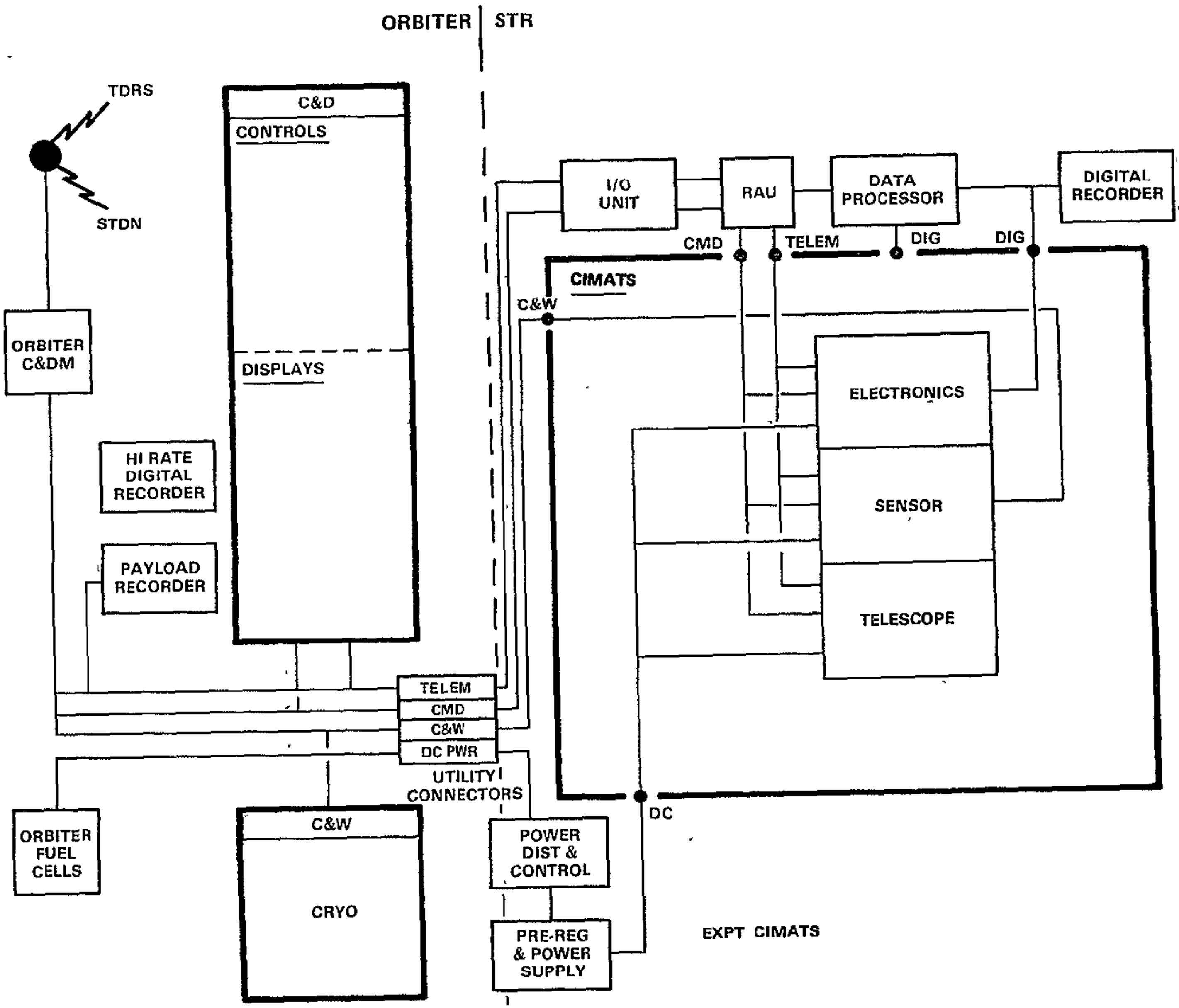


Figure 3-15. EVAL Experiment Schematic - CIMATS

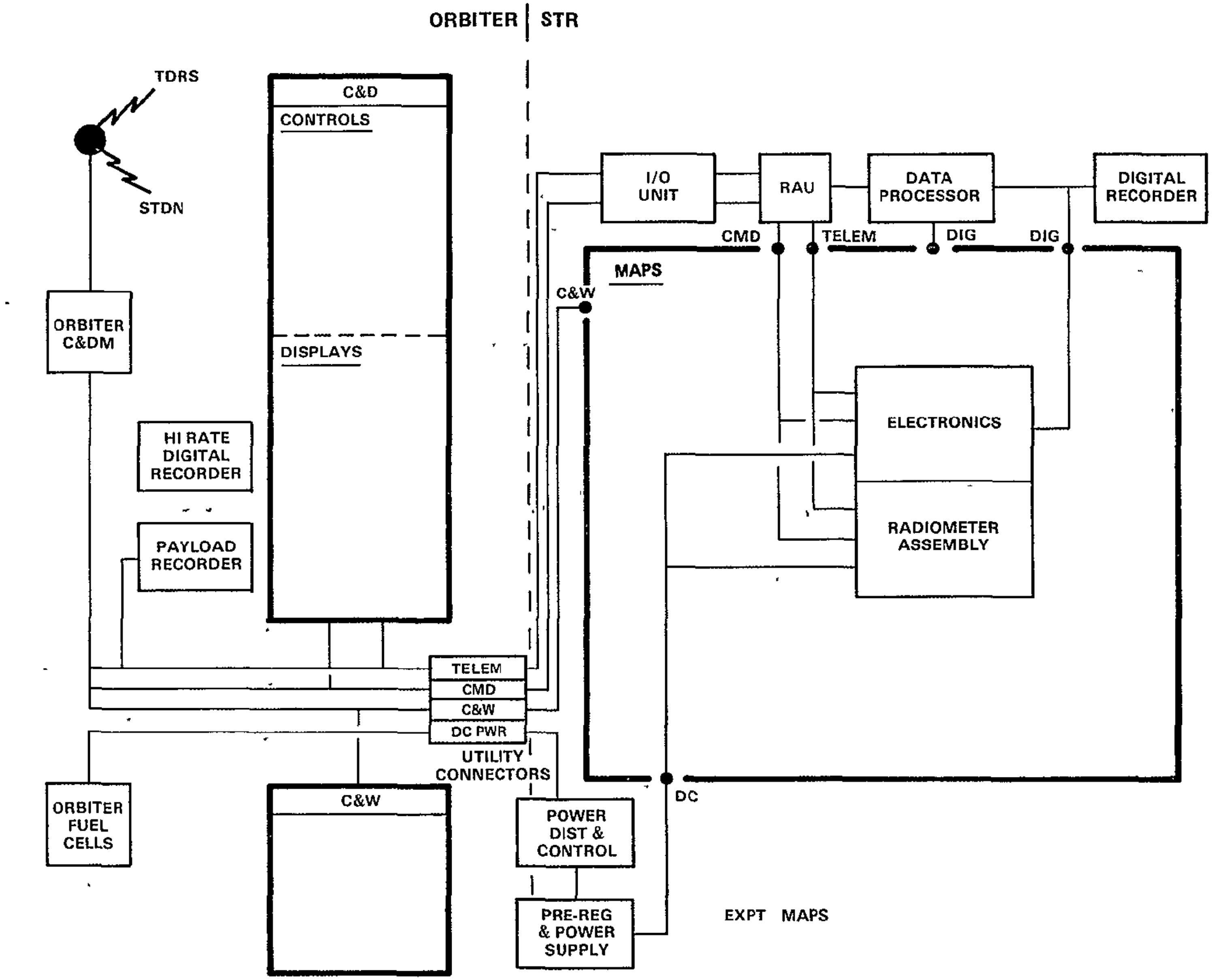


Figure 3-16. EVAL Experiment Schematic - MAPS

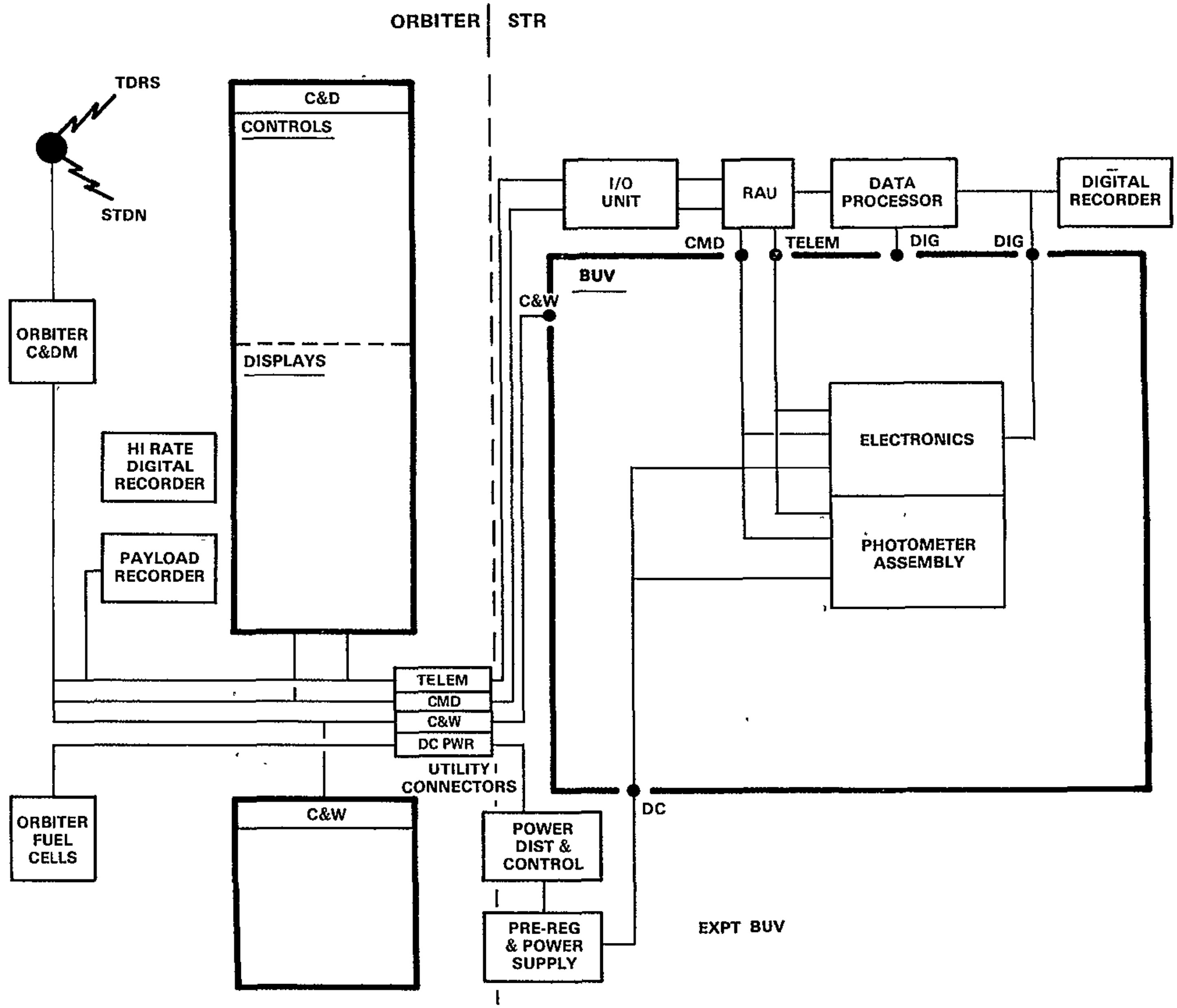


Figure 3-17. EVAL Experiment Schematic - BUV

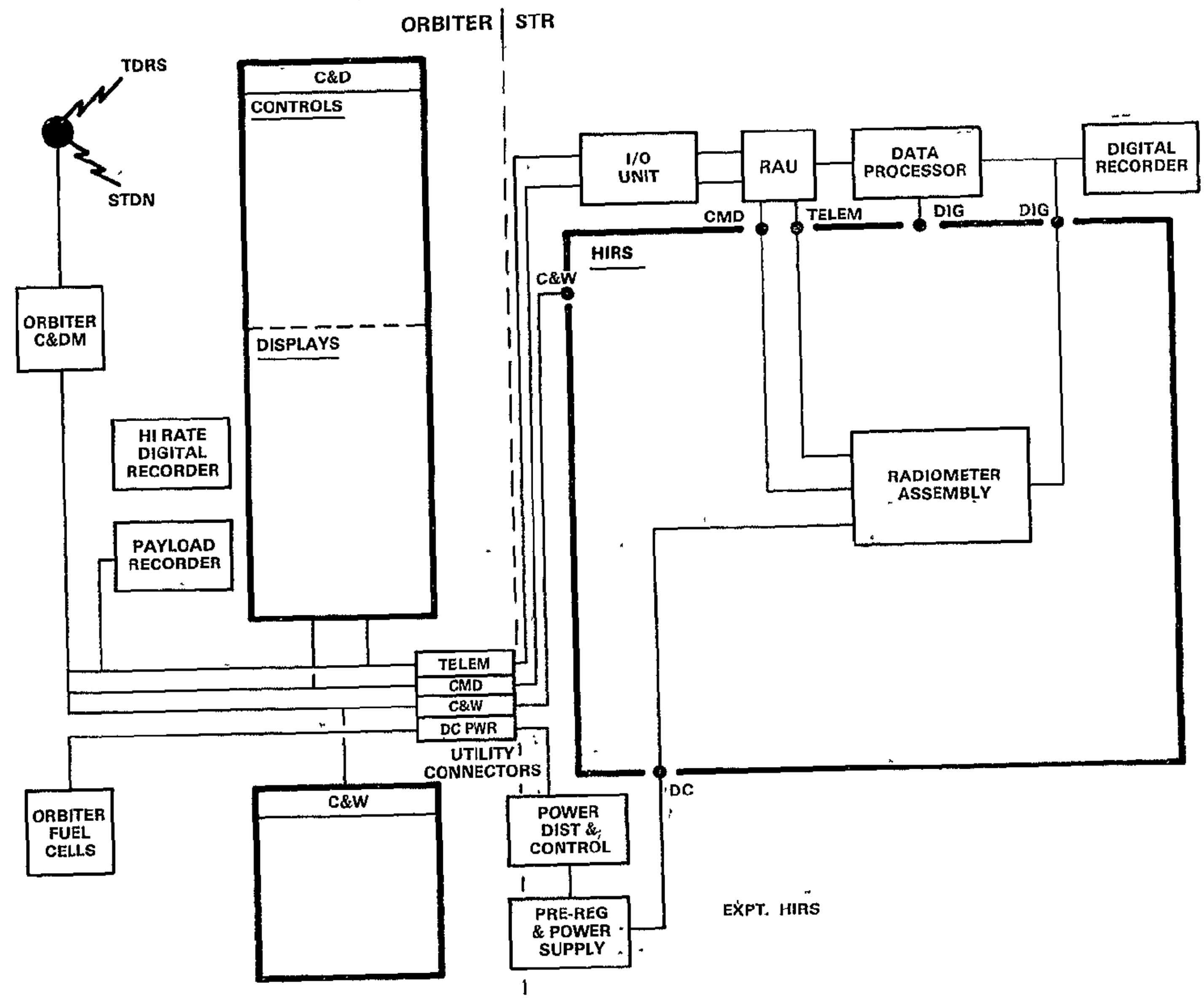


Figure 3-18. EVAL Experiment Schematic – HIRS

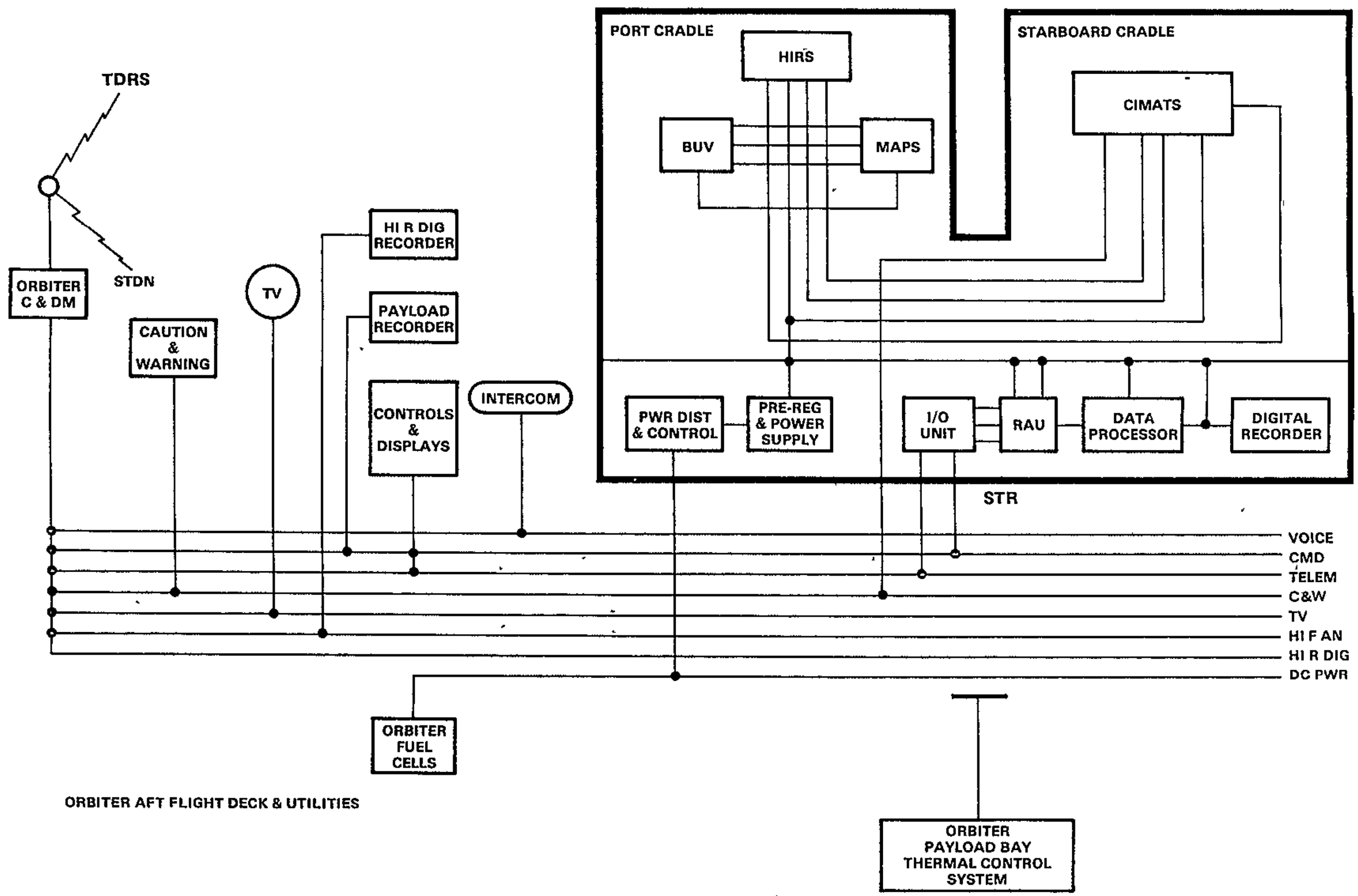


Figure 3-19. EVAL/STR System Schematic

3.4 OPERATIONS

3.4.1 EXPERIMENT OBSERVATIONS

An assessment of on-orbit mission operations related to the STR payload has been performed to determine operational timeline and resource profiles. The timelines only pertain to experiment operation, and not crew requirements; since the use of STR implies essentially automated operation. Similarly, the only resource profile of concern involves power. STR utilizes power supplied by Shuttle, and regulator and distributes it to the various sensors and its own subsystems. Consequently, knowledge of the STR payload power profile is pertinent. Conversely, STR employs self-contained data storage capabilities; therefore time sharing of STDN or TDRS data links is not required.

The approach used in this analysis to determine the operational requirements for the STR payload assumed that the deployment of the two spacecraft and the retrieval of the third spacecraft occurs in the initial phases of the mission with the remainder of the mission utilized to satisfy the STR mounted experiment requirements.

Orbital ground traces were generated for a 7 day sortie mission having the specified conditions of 460 km altitude, 33° inclination and assuming a launch from the ETR at Cape Kennedy. Orbit decay rate and eccentricity were both assumed to be zero with orbit injection assumed to be over Cape Kennedy at the time of launch for simplicity. It was also assumed that launch occurred at 0700 EST on October 15, 1981.

From the orbit calculations, ground tracks were obtained which indicate the global coverage available to the STR experiment package. A sample of this coverage for a typical one day time frame, approximately 16 orbits, is shown in Figure 3-20. Since experiment operation is not predicated on specific geographic locations, but rather local lighting conditions, specific operational scheduling is covered in the mission time line section. A synopsis of the total data gathering times for each experiment was:

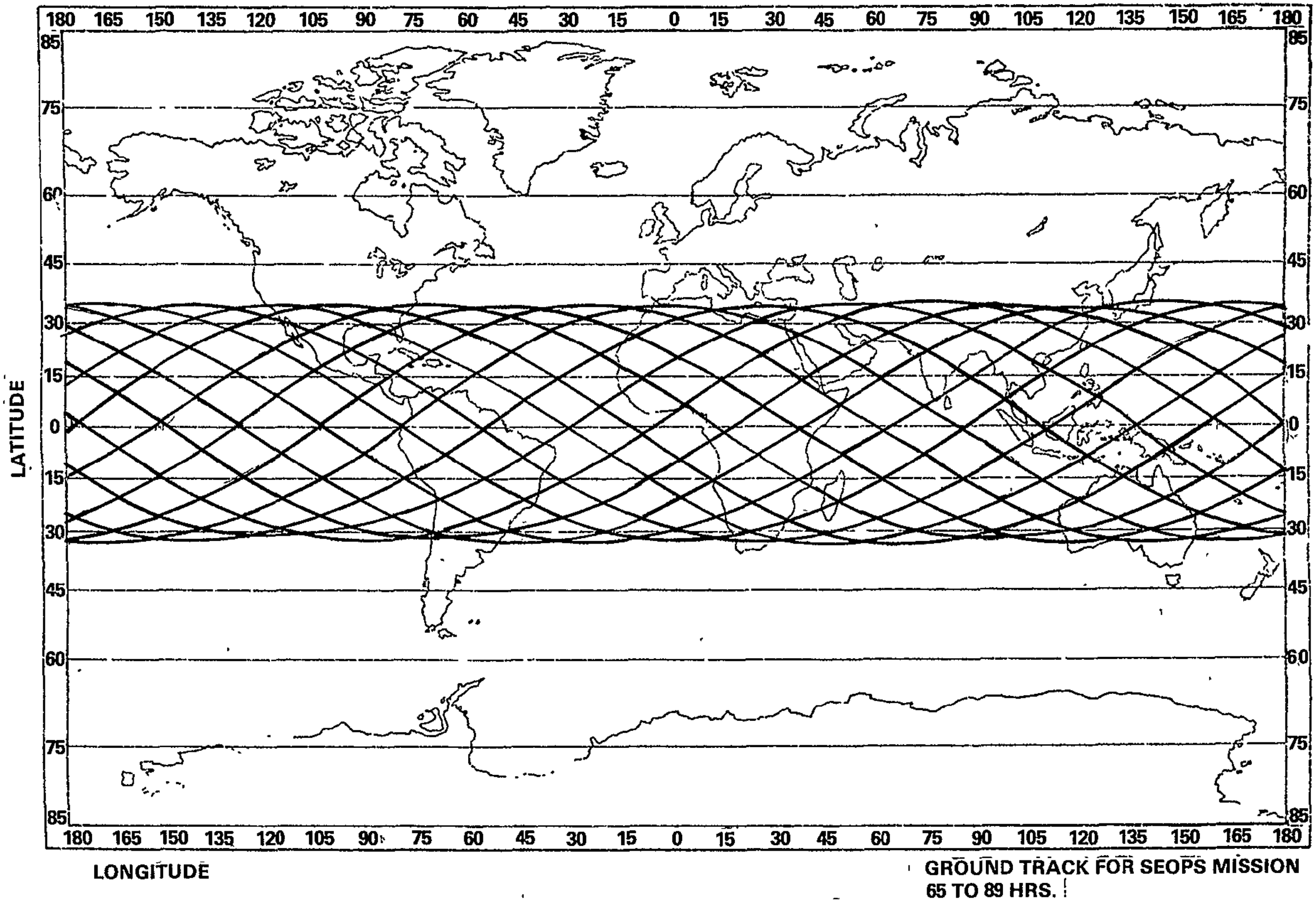


Figure 3-20. STR Mission Ground Trace

CIMATS	-	5731 min
HIRS	-	5731 min
MAPS	-	1594 min
SBUV	-	1594 min + 62 min allocated to sun calibrations

Normal experiment operations were terminated at an elapsed mission time of 155 hours to allow for crew rest and reentry preparations for a scheduled deorbit and landing on orbit 109 at approximately 0800 of the seventh day.

To determine the start of the STR experiment operations, it was necessary to determine the time involved for payload satellite preps and predeployment tests required for orbital deployment, in addition to retrieval operations for the orbiting satellite.

The following are the time allocations assumed necessary for satellite deployment and retrieval operations.

- Launch through orbiter systems check out - 4.5 hours
- Very Long Baseline Interferometer Spacecraft
  - Spacecraft test & deployment - 13 hours
  - Crew eat periods - 3.0 hours
  - Crew rest & sleep periods - 9.5 hours
- Gravity Probe "B" Spacecraft
  - Spacecraft test & deployment - 13 hours
  - Crew eat periods - 4.0 hours
  - Crew rest and sleep periods - 9.5 hours
- Solar Max Spacecraft
  - Spacecraft retrieval - 3.0 hours

  Total elapsed time - 59.5 hours

3.4.2 MISSION TIMELINES

The following experiment constraints were the only considerations assessed necessary for developing a Shuttle compatible timeline.

- Correlation Interferometry for Measurement of Atmospheric Trace Species (CIMATS)
  - Full global coverage; no lighting restrictions
- High Resolution IR Spectrometer (HIRS)
  - Full global coverage; no lighting restrictions
- Monitoring of Air Polution from Satellites (MAPS)
  - Full global coverage; prefered local ground sun lighting angle of 25° or greater
- Solar Backscatter UV Spectrometer (SBUV)
  - Full global coverage; prefered local ground sun lighting angle of 25° or greater; one sun calibration per orbit

Since the CIMATS and HIRS experiments did not require any considerations relative to orbital position, these experiments were turned on at 59.5 hours of mission elapsed time and continued operating to the 155 hour mission elapsed time-experiment termination time. The MAPS experiment was scheduled for data taking sequences when the nadir ground sun angle was 25° or greater. The SBUV experiment was scheduled for a solar calibration during the Shuttle sunrise period, with the experiment remaining in the operating mode through the ground sun angle zone of 25° or greater. A sample of one day's mission timeline is provided in Figure 3-21. The power profile shown across the bottom of the timeline indicates the power profile for the STR sensors only. The mission "on" times are indicated by the horizontal dark lines, while the interval encompassed by the vertical tick marks on these lines denotes the actual data gathering period.

It was assumed that vehicle maneuvers involved in predeployment satellite tests and satellite retrieval operations were not compatible with the earth viewing requirements of the STR experiments. The constraint precludes scheduling during the satellite operations. Also, other than periodic status checks, crew involvement is not anticipated in normal STR operations.

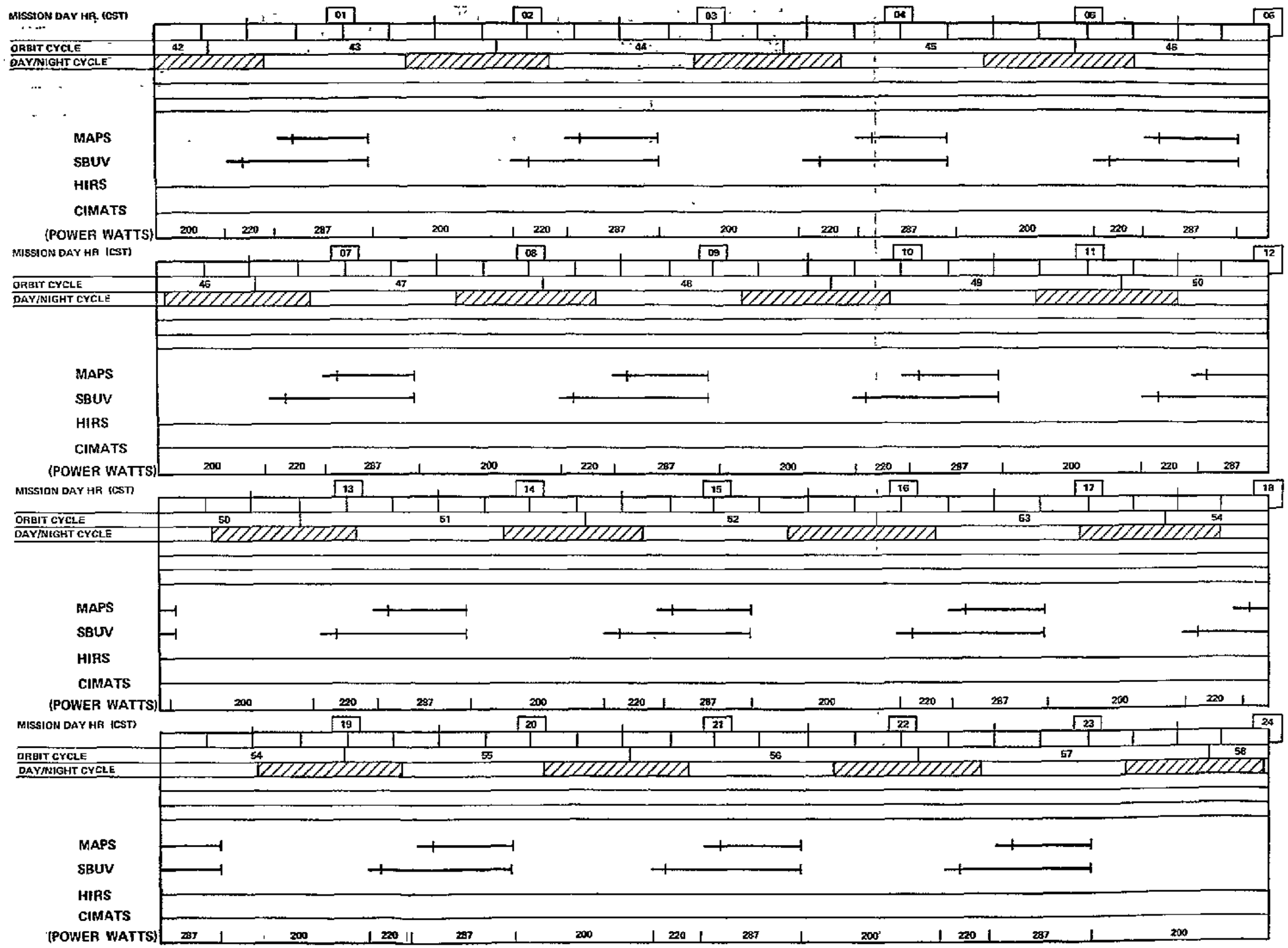


Figure 3-21. Sample Mission Time Line

FOLDOUT FRAME 1

FOLDOUT FRAME 2

3.4.3 GROUND OPERATIONS

Physical integration of Shuttle payloads prior to flight will involve four levels of activity, beginning with checkout of payload subassemblies (Level IV) and ending with installation of cargo into the Orbiter (Level I). The precise definition of each integration level has shown some amount of evolution as Shuttle era concepts have matured. As of November, 1976, the levels of integration as they apply to EVAL payloads are as follows:

1. Level IV. Integration and checkout of EVAL equipment assemblies with individual Spacelab racks on pallets segments, or with the STR bridge/cradle. Since the typical EVAL experiments includes equipment mounted in more than one location, a major precursor effort is required to configure, assemble, checkout, and generally prepare the experiment equipment for Level IV integration.

2. Level III. Combination, integration, and checkout of all experiment/Payload mounting elements (e.g., Spacelab racks and pallets segments, STR bridge (cradle) with experiment/payload equipment already installed; and of experiment/payload and Spacelab software. Here Spacelab racks and/or pallet segments containing EVAL and other payload equipment are assembled into their flight configuration and checked out as a system. STR is included in this checkout when it is an integral part of the Spacelab payload.

3. Level II. Integration and checkout of the combined experiment/payload equipment and their mounting elements (e.g., Spacelab racks and/or pallet segments, STR bridge/cradle) with flight support elements (e.g., Spacelab module segments or igloo).

4. Level I. Integration and checkout of total cargo (Spacelab modules and/or pallets and STR plus any other payload such as automated spacecraft or piggyback packages) with the Shuttle orbiter, including the necessary pre-installation assembly and testing with simulated interfaces.

The current baseline is that Levels I, II, and III integration will be at KSC while Level IV will take place at other sites. For EVAL, Level IV integration will probably be performed at the site of an integration contractor along with most pre-Level IV assembly and test of experiment equipment. Other payloads in shared EVAL Missions may be integrated elsewhere and need not meet the EVAL payloads until both arrive at KSC for Level III integration. The precise interface between Level IV and Level III integration is presently under re-evaluation and is consequently unclear, but for this study Level IV integration is assumed to end with validation of individual Spacelab racks and pallet segments and the STR bridge/cradle.

The EVAL/ STR payload is essentially ready for Level I integration upon arrival at KSC (after Level IV integration). The only Level III/II tasks to be performed are checkout of Payload Specialist Station (PSS) hardware and software in the Operations and Checkout (O&C) Building workstands prior to transfer to the Orbiter Processing Facility (OPF) for cargo integration and installation. It is possible that PSS modules and other equipment from the primary deployment and retrieval payloads will cycle through the O&C Building workstands for Level II (multiple payload) integration. EVAL/ STR preflight integration activities are summarized in Figure 3-22.

Principal Investigators (PIs) and/or their teams will be expected to participate in all levels of physical integration, actively supporting EVAL equipment installation, integration, and performance testing. They will participate in mission simulation exercises, where on-orbit and ground support personnel rehearse their respective roles. In the case of EVAL/STR payload, the involvement of the PI and/or his team during KSC operations and flight is minimal.

Post-flight activities begin as soon as the orbiter has landed and has been towed to the OPF. The payload bay doors are opened approximately 15 hours after landing and payload removal in completed in a single work shift. The SMM spacecraft must be removed before the STR cradle can be removed. After removal, the EVAL/STR payload is transported to the O&C Building and made ready for shipment to the EVAL integration site, where payload de-integration and data stripout is accomplished. Post-flight activities are summarized in Figure 3-23.

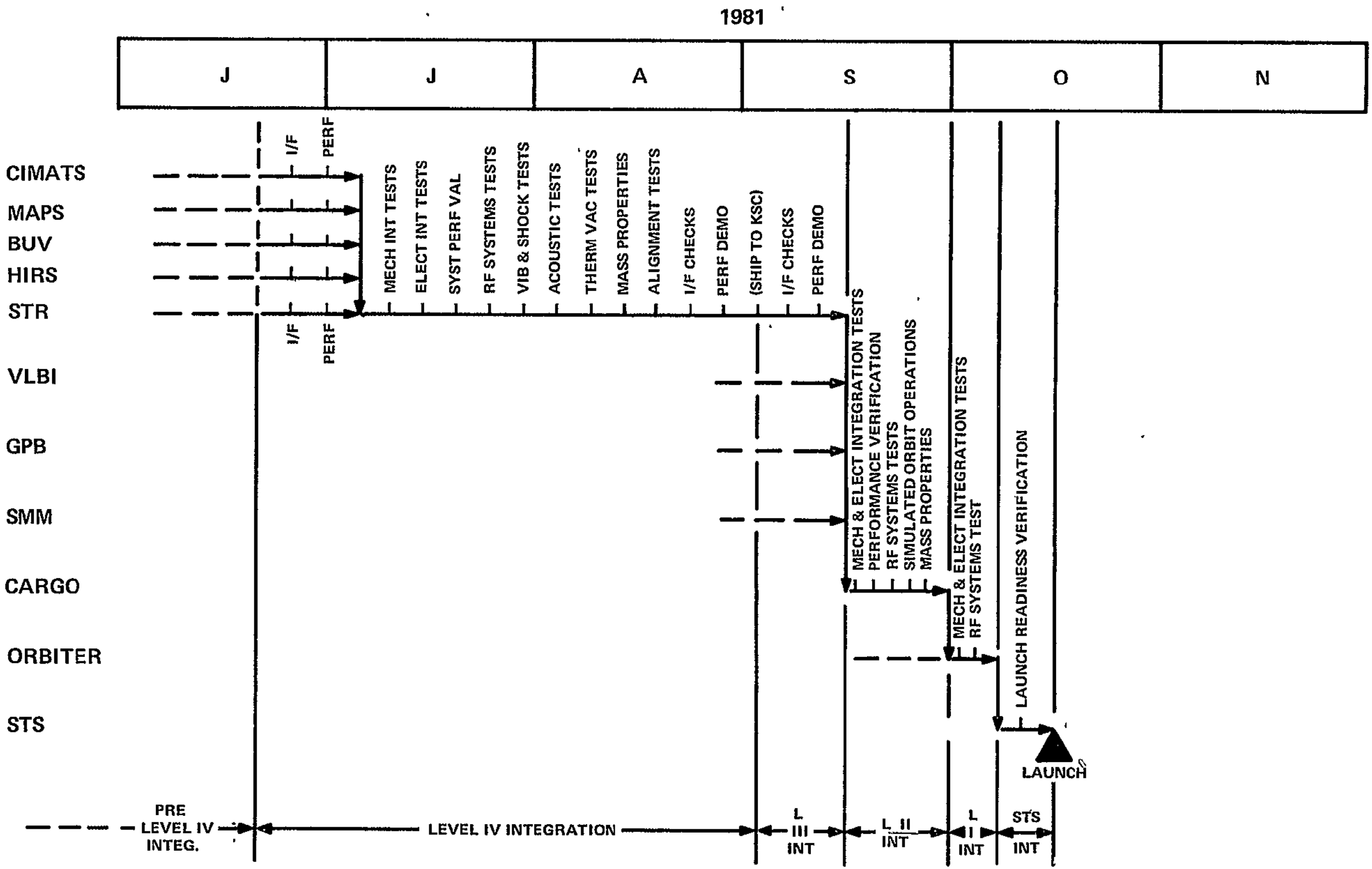


Figure 3-22. EVAL/STR Preflight Integration

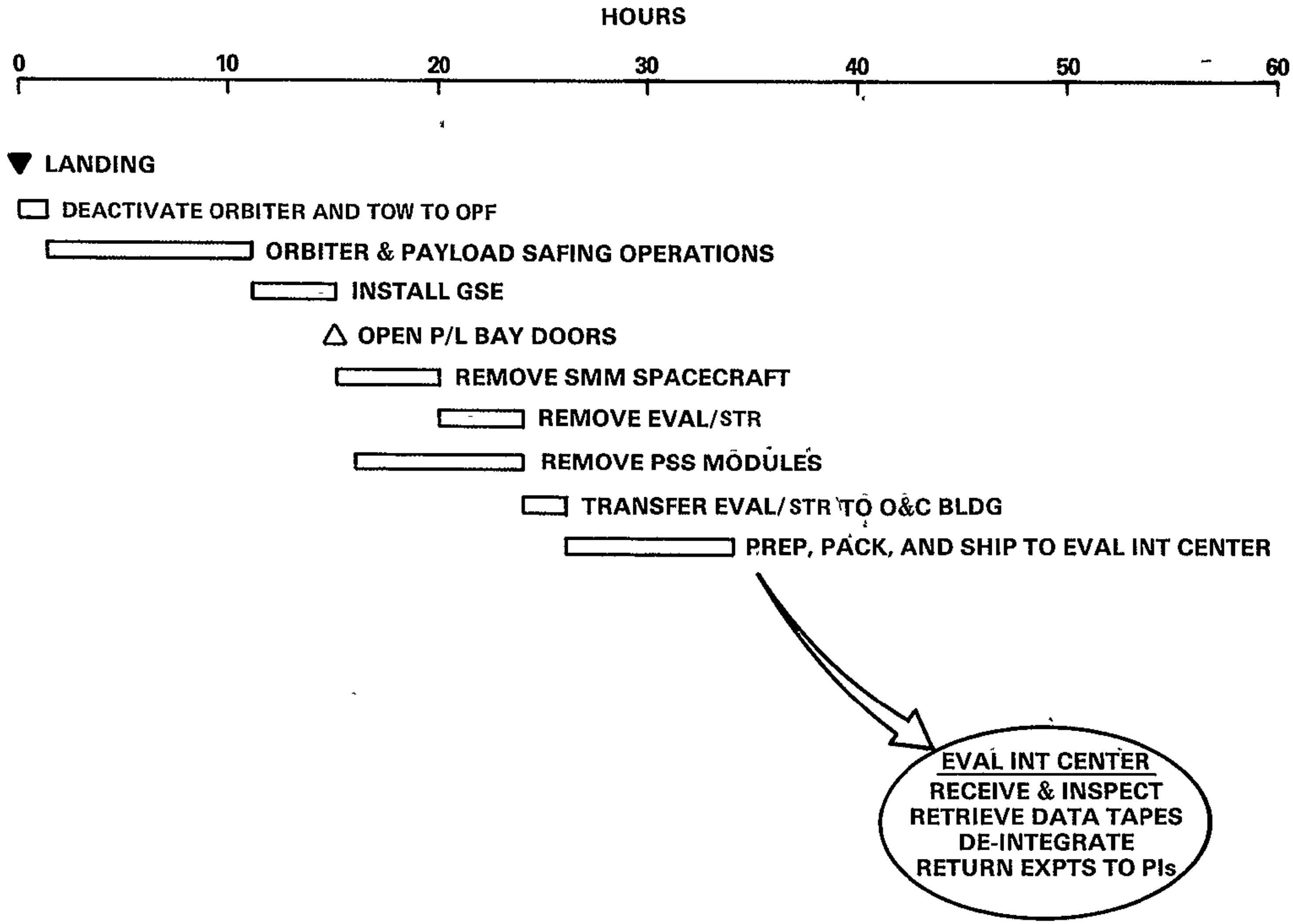


Figure 3-23. EVAL/STR Post Flight Activities

3.5 SYSTEM REQUIREMENTS AND ACCOMMODATIONS

The STR system is the physical interface with the Environmental Quality sensors and provides most of the support subsystems required for this payload. The Shuttle Orbiter, however, does provide the input power for STR; and since the pointing and stability requirements are modest for this payload, the Shuttle Orbiter also provides this capability.

3.5.1 POWER/THERMAL

STR will interface with the Shuttle through a cable harness from the Orbiter's DC-2 bus. The Orbiter will provide a STR dedicated power connector and will not interface with the rest of the Shuttle payloads. The input voltage to STR is 24 $\pm$ 4 Vdc; all other voltages, power distribution, EMC filtering, as well as overvoltage protection will be performed by the STR power conditioning subsystem (Figure 3-24).

The power distribution box accepts the incoming raw dc power and provides control relays to turn on and off the primary power to the voltage preregulator. It also serves other functions for housekeeping purposes. The dc/dc converter provides regulated housekeeping voltages such as $\pm$ 15 V.

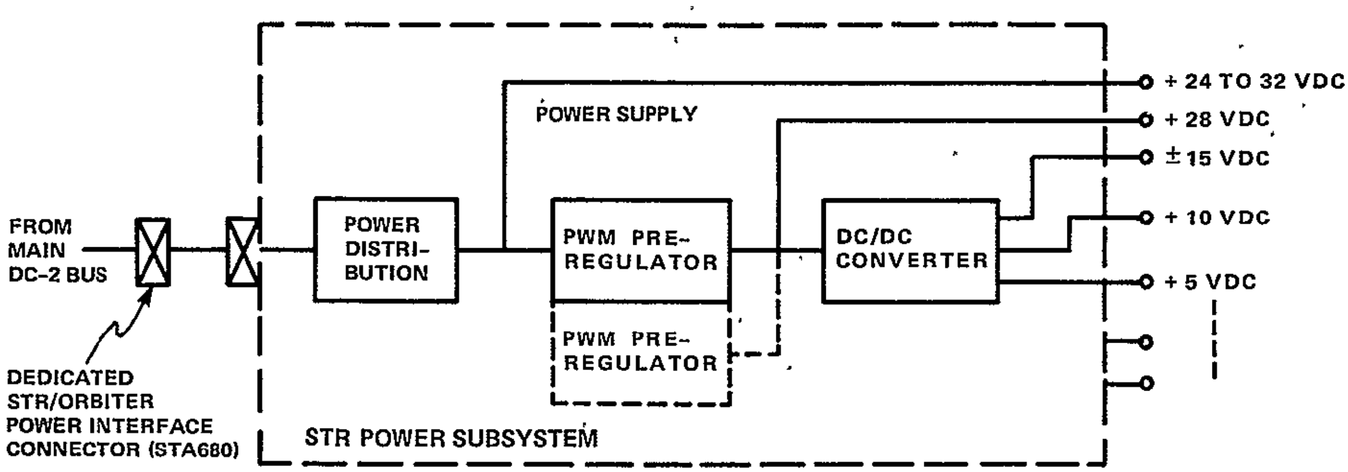


Figure 3-24. STR Power Subsystem

The maximum power requirement from the Orbiter for the four instruments is 290 watts, with approximately 100 watts of additional power for the support subsystems (Table 3-7). Therefore the total maximum power consumption will not exceed 400 watts. Figure 3-25 shows the mission power timelines.

Table 3-7. Power Consumption

Component	Power Consumption
Power Distribution	5W
P. W. M. Regulator (500W)	50W.
DC/DC Converter (100W)	15W
CMD/TLM & Data Handling	15W
RAU	1W
10^9 Recorder	15W

The energy consumption is 22.2 kwh for the four instruments with an additional 5.5 kwh for the support subsystems, totaling 27.7 kwh of energy required from the Space Shuttle. The DC-2 bus has a 50 kwh energy capacity which is more than adequate for this experiment.

The sensors hard mounted to the STR structure all dissipate relatively small quantities of power (20 to 180 watts). Thus, STR can reject the heat load using its own louver thermal subsystem. STR will be able to maintain component surface temperatures between a maximum average of 21°C and minimum of 5°C, and at the same time be thermally isolated from the Shuttle and independent of the Shuttle thermal support.

3.5.2 POINTING & ATTITUDE CONTROL

The Space Shuttle has the capability of providing pointing to an accuracy of $\pm$ 0.5° for earth viewing missions. This pointing accuracy can degrade for about $\pm$ 2.0° for payloads located in the payload bay due to structural flexture of the Space Shuttle. For the EVAL Environmental Quality mission, Shuttle pointing support and the maintenance of an attitude of $\pm$ 0.01 deg/sec/axis is adequate since each one of the four instruments requires only a general knowledge of the Shuttle pointing direction.

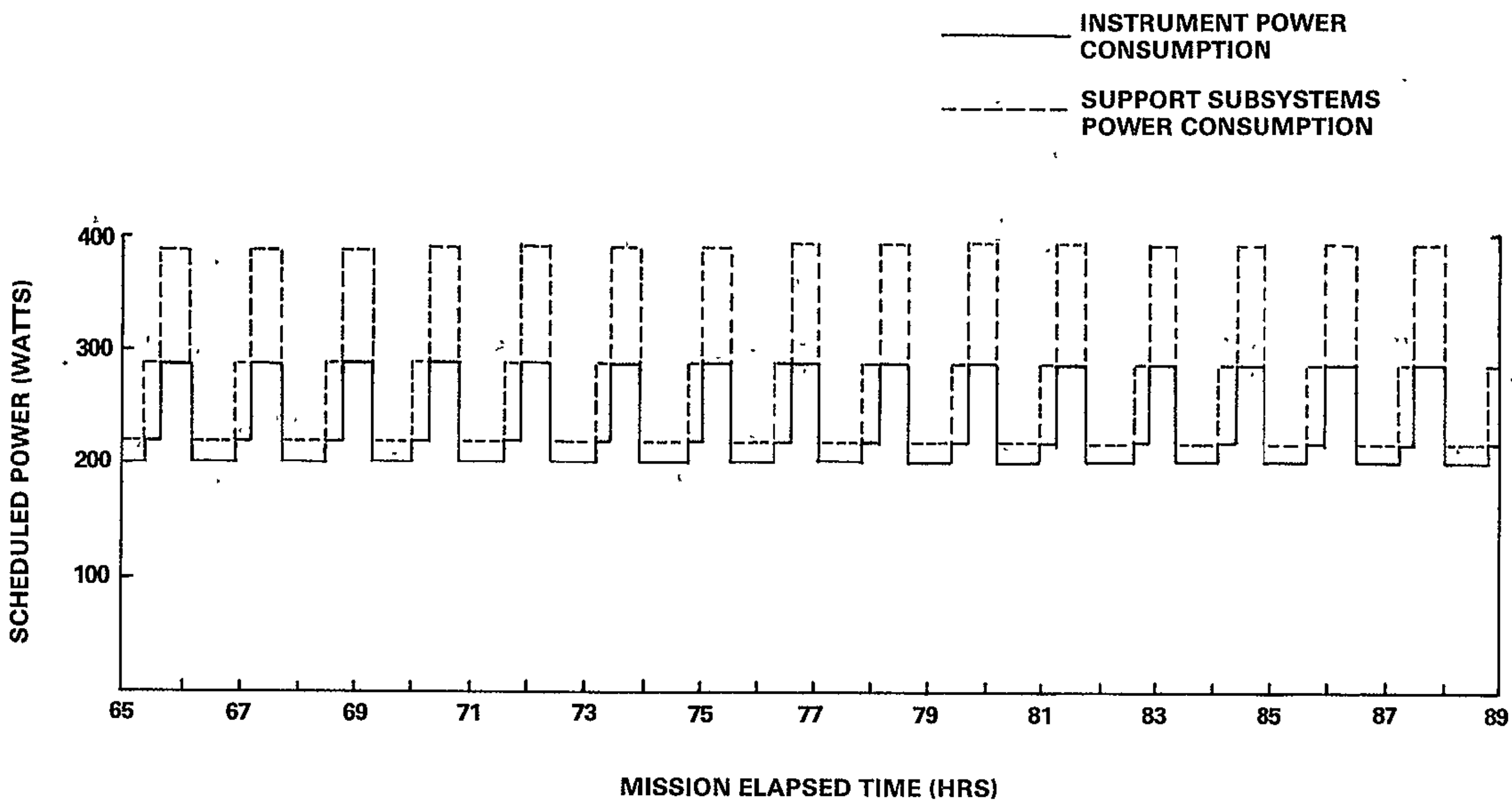


Figure 3-25. Mission Power Timeline

The four instruments will be hard mounted on the STR structure. Initially the instrument line of sight will be aligned with the Space Shuttle and utilize the Shuttle as a pointing platform. The velocity vector of the Shuttle will be in the direction of the length of the cargo bay.

Shuttle attitude information will be used to start the experiment and will be recorded as useful data throughout operation of the Environmental Quality experiment.

3.5.3 COMMAND, CONTROL AND DATA

The four STR/EVAL experiments all have very low data rates, ranging from 220 Bps for the SBUV to 3369 Bps for the HIRS. All four instruments will be commanded with pre-stored commands on STR and periodically updated by the Shuttle attitude information.

A NASA standard 10^9 bits recorder (Table 3-8) provided by STR is used to record all of the experiment data as well as all housekeeping functions.

Table 3-8. Specifications for Standard NASA Recorder

Parameters	Specification
Capacity (bits)	3×10^9
Data Tracks	14
Maximum Bit Rate, bps	120,000
Record to Playback Ratios	120:1 to 1:120
Minimum Record to Playback Time	3.73 minutes
Maximum Record to Playback Time	160 hours
Power (depends on speed)	10 to 30 watts
Weight	28 lbs.

A total of 95.5 hours of operation will require a maximum of about 2.5×10^9 bits of storage capacity.

There is no need to transmit the data from the Space Shuttle since the information will be used for technique analysis or application development, and real time evaluation of the data is not required. This approach requires minimum Shuttle interface and is the simplest and most cost effective.

Telemetry information will be displayed on the mission specialist's CRT for monitoring purposes if desired and will display go-no go conditions with command override capability by the mission specialist.

SECTION 4

SENSOR MODIFICATIONS

The EVAL payloads analyzed to date (a partial Spacelab, a full Spacelab, and a STR)* have all been comprised of multiple sensors: frequently the same sensor has been used on more than one of the payloads. Table 4-1 lists the various sensors included in each payload.

Table 4-1. EVAL Payload Sensors

Partial Spacelab Payload	Fully Spacelab Payload	STR Payload
TM	SIMS	CIMATS
LFC	SIR	HIRS
S-193	TM	MAPS
ALT	LFC	SBUV
SMMR	ALT	
LRS	S-193	
CPR	LRS	
LACATE	CPR	
HALOE	SBUV	
SER/SAGE	AMPA	
HSI		
SBUV		
ESP		
EEE		

These sensors basically represent two different heritages: those which have been/are being developed primarily for application on free flying satellites; and those which are initially being designed for Shuttle applications. The sensors in the latter category are being developed with full consideration for the environmental and operational characteristics associated with Shuttle; and can be integrated into payloads in their original design configuration. Those sensors which have been/are being designed with another carrier primarily in mind, however, will generally have to undergo some degree of modification to be

*The partial Spacelab payload is documented in GE Report No. 76SDS4269, EVAL Concept Definition - Partial Spacelab Payload, dated 30 September 1976; while the full Spacelab and STR payloads have been described in previous sections of this report.

compatible with Shuttle sortie applications. Table 4-2 identifies those EVAL payload sensors which will require modification.

Table 4-2. Identification of Sensors Requiring Modification

Sensors Requiring Modifications	Sensors Designed As Shuttle Payloads
LACATE	SIMS
SER/SAGE	SIR
ESP	AMPA
SMMR	EEE
LRS	HALOE
CPR	HSI
S-193	
ALT	
SBUV	
LFC	
TM	
CIMATS	
HIRS	
MAPS	

Table 4-2 indicates that more than twice as many sensors (14 out of 20 considered for the EVAL payloads) will require modification as not. This fact is synonomous with cost effective payload development; since modifications to existing sensors, where possible, is generally accomplished for fewer dollars and shorter times than creating new designs.

The purpose of this section is to identify on a preliminary basis those modifications which are required for the specific sensors identified in Table 4-2, and provide an estimate of the time and dollars required to affect the modification.

4.1 LOWER ATMOSPHERE COMPOSITION AND TEMPERATURE EXPERIMENT (LACATE)

The LACATE sensor is a multispectral infrared scanning radiometer that scans the IR emission from the earth's horizon and produces radiance profiles in nine spectral bands between 6.2 and 17.5 microns. The spectrometer instrument is approximately 92 cm. high by 36 cm. in diameter, and weighs 22 kg. Figure 4-1 shows a cross sectional view of the sensor. The electronics has an additional weight of 7 kg.

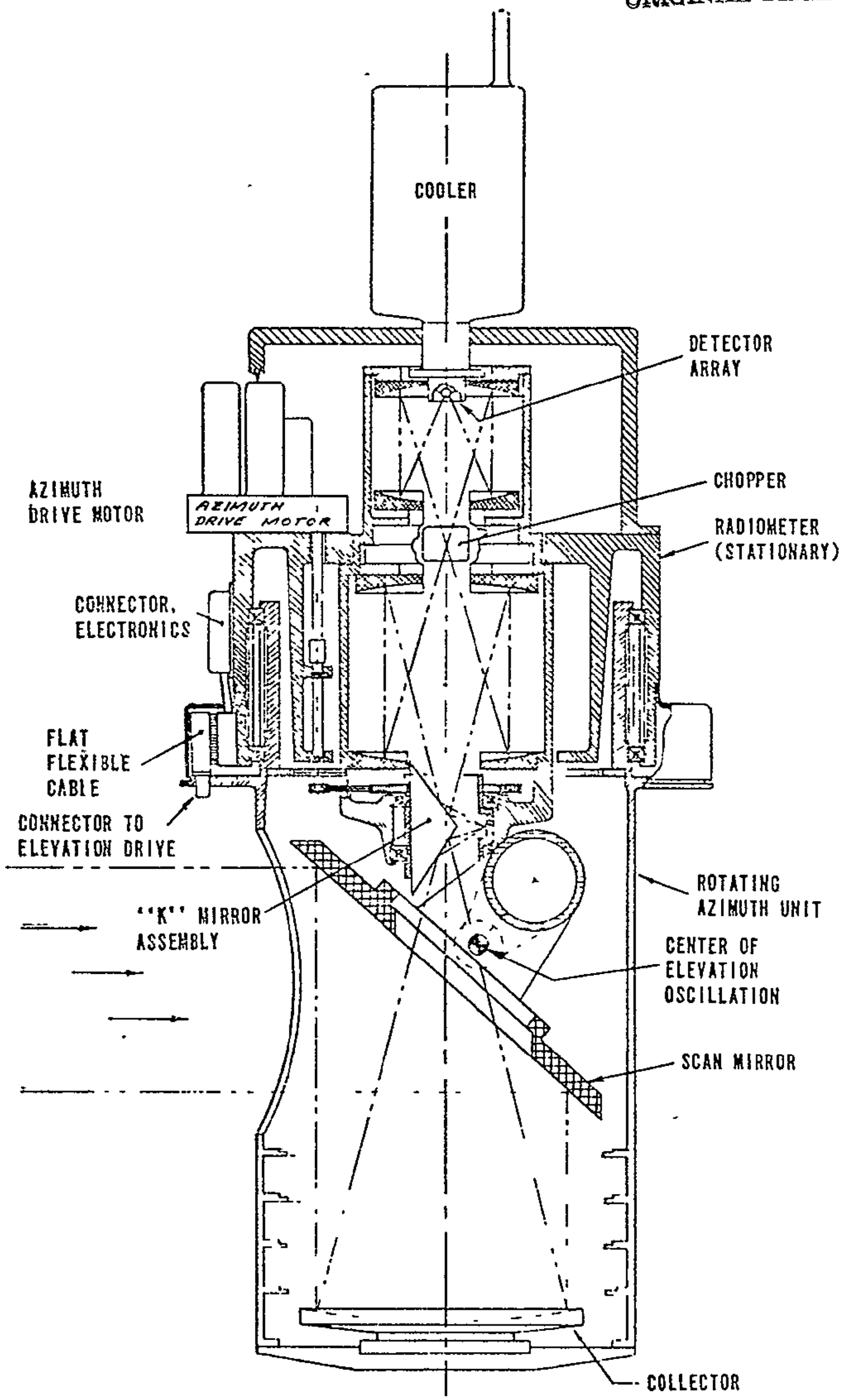


Figure 4-1. LACATE Radiometer

The sensor measures temperature vertical distribution and the concentration of O_2, H_2O, CH_4, HNO_3, NO_2, & N_2O from the upper troposphere to the middle stratosphere.

4.1.1 MODIFICATIONS

The current AAFE instrument will require the following modifications and adaptations to be compatible with the EVAL facility on Spacelab.

Structural

The sensor housing requires reinforcement and acoustic protection to be able to withstand the launch loads. The original housing was lightened for the balloon mission by removal of material from the sensor housing. The combination of static load, mechanically induced vibration, and acoustically induced vibration may be excessive.

Acoustic protection for the housing and internal components consists of noise dampening and absorption in the form of an enclosure made of visco-elastic epoxy material between sheets of aluminum. Mechanically induced vibration can be attenuated through specially de-tuned mounts at the sensor to test rack interface.

The azimuth drive assembly should be redesigned and rebuilt to eliminate the present bearing malfunction.

Optical

Optical elements should be removed and cleaned. The "K" mirror must be recoated to eliminate polarization effects. The optical magnification must be analyzed to determine means to improve the signal modulation. As a result of the analysis, the chopper blades can be repositioned.

The elevation scan mirror drive mechanism was designed for an angular travel suitable for the high altitude balloon experiment. The EVAL orbital parameters will require a new range of elevation angles to permit a complete vertical profile through the atmosphere.

Thermal

For the balloon-borne experiments, the LACATE detector is cooled with an open-loop cryogenic system using a liquid nitrogen dewar. In order to accommodate the longer duration of the EVAL mission, two alternatives need to be analyzed: (1) use of a larger dewar and liquid nitrogen dispensing system; (2) use of a closed-cycle cooler.

The overall LACATE instrument must be maintained at ambient temperature limits, therefore the acoustic enclosure should be thermally protected with super-insulation and a radiation surface must be provided to dissipate the generated heat. Detailed thermal analysis will show whether or not one or two sides of the enclosure will be used as radiators or whether an external radiator should be used. Preliminary analysis shows that heat pipes will be required to transfer the heat to the radiating surfaces.

Electrical

A space rated power conditioning unit should be designed and purchased for the instrument. Complete electronics repackaging will be necessary for the Spacelab flight to ensure survivability after exposure to the launch and space environment, and to accommodate modifications to the electronics. New harnesses will be required including Shuttle compatible harnesses.

Command/Control

The scan mirror servo loop requires modification to: (1) increase the response of servo loop stiffness to compensate for possible degradation in mirror assembly performance; (2) become less susceptible to EMI by up-grading the methods of grounding and shielding.

The instrument electronics signal level and format must be made compatible with the command system which employs the Spacelab C&DMS computer and a command decoder which is shared among several experiment instruments.

Data Management

The instrument telemetry signal conditioning circuits must be made compatible with the EVAL/Spacelab data handling system. The interface circuitry should consider the following functions, depending on the EVAL data management approach selected:

1. A to D conversion
2. Sub-multiplexing
3. Input to RAU
4. Input to High Rate Data Multiplexer

Contamination

An actuated cover will be required at the thermo-acoustic enclosure to permit protection of the optics during launch operations, ascent, and the early part of the orbital phase.

Pointing and Stabilization

No gimbal system will be required since the instrument contains a servo driven pointing mirror.

4.1.2 MODIFICATION COST AND SCHEDULE

Rough Order of Measure (ROM) Cost estimate: $500K to $760K dependent upon the options implemented

Duration: 18 months

This cost and schedule estimate is based on inputs obtained through NASA/LRC and documented in the "Locate Balloon Radiometer Refurbishment Analysis" prepared by Honeywell's Radiation Center. Included in the estimates are the modification analysis and design, rework and refurbishment, and purchase of new components required for adaptation to Shuttle missions.

4.2 STRATOSPHERIC AEROSOL AND GAS EXPERIMENT (SAGE)

SAGE is a radiometer that measures stratospheric aerosols, ozone, and NO_2 as a function of altitude, latitude and longitude. The instrument permits measurements of the sun's intensity through the earth's limb at four spectral intervals: namely 1.0, 0.6, 0.45 and 0.38

micrometer spectrum during the spacecraft's sunrise and sunset. The instrument will scan the sun with an instantaneous field-of-view designed to produce 1 Km vertical resolution at the atmosphere.

The telescope employed is a 5 cm Cassegrain F/30 with a 3.56 cm diameter central obscuration. The total radiometer error budget allows an error of 0.1% of full scale.

Figure 4-2 shows a cross sectional view of the instrument.

4.2.1 MODIFICATIONS

The instrument has been designed for the Atmospheric Explorer Module and requires few modifications to make it compatible with Shuttle missions.

Structural

No structural changes are required except for the possible re-inforcement of the circuits in the electronic package to withstand the Shuttle dynamic environment. A mount system will be required to reduce mechanically transmitted vibration, particularly at the resonant frequencies of the supporting structure.

Optical

The current instrument uses a scanning mirror with a depression angle range from 13^{o} to 29^{o}, based on an orbital altitude of 1110 km. The lower altitude and different inclination of the EVAL mission will require a change in the mirror elevation drive mechanism.

Thermal

Thermal balankets should be redesigned to conform with the installation of SAGE on EVAL. It is anticipated that new heaters will be required to maintain the temperature of the instrument during the night portion of the orbit.

Electrical

No modifications are required, since the instrument specifications call for satisfactory operation with a +28 VDC $\pm$ 4 VDC unregulated power source.

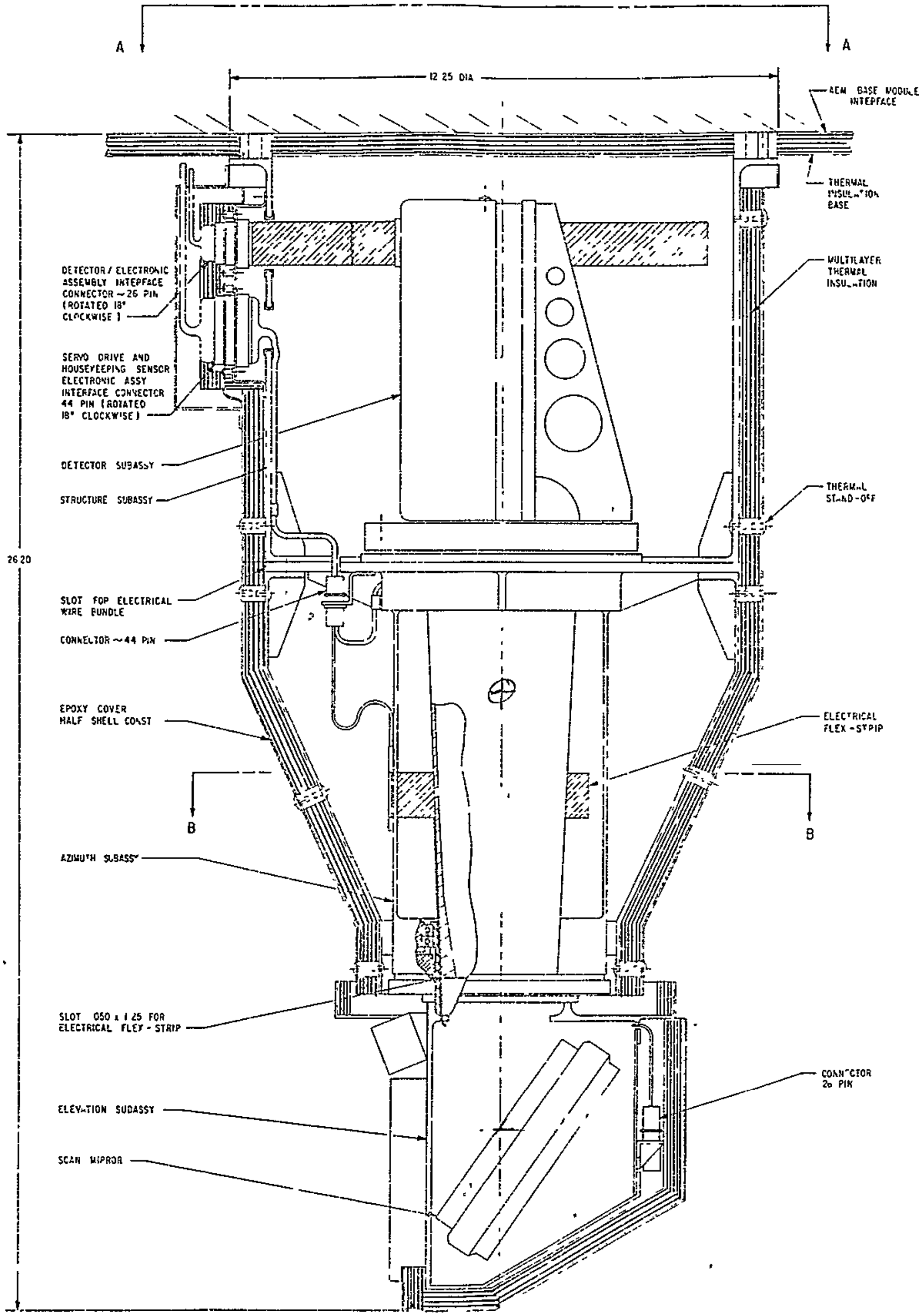


Figure 4-2. SAGE Radiometer

Command/Control

Command tests should be performed to determine the compatibility of the instrument to stimulated command signals from the Spacelab/EVAL Command System. Interface circuitry can be defined from the results of these tests.

Data Management

The data system used in conjunction with the AEM uses a 12 bit record digital system. In addition to digital data, the instrument requires transmission of 8 samples per second of analog data. Interface circuitry will be required to establish compatibility between the instrument and the Spacelab/EVAL data handling system.

Contamination

The optics are susceptible to degradation due to molecular and particulate deposition on the mirror and telescope optics. A remotely actuated cover should be incorporated at the optical inlet port to prevent contamination during launch operations, ascent, initial Shuttle System outgassing in orbit, and during return operations.

Pointing and Stabilization

The internal pointing capability within the instrument is capable of pointing within $\pm$ 180° to $\pm$ 2 arc minute accuracy in azimuth, and 13° to 29° depression angles to $\pm$ 30 arc seconds in accuracy. The platform upon which the instrument is mounted must face local vertical within an accuracy of 1° in pitch and roll and 2° in yaw. Knowledge of pointing is required within 0.5 degrees in all axis. Attainment of the desired sensor platform pointing accuracy and knowledge of pointing requirements will necessitate an attitude reference system mounted near the instrument, and used in either of two ways:

(1) the attitude error signal connected to the Shuttle orbiter attitude control servo loop,

(2) the attitude error signal connected to the SAGE azimuth and elevation control servos

The latter alternative is the more desirable one, however, in EVAL missions where many instruments require improved pointing accuracy and stability from the orbiter, alternative (1) may be more practical.

4.2.2 MODIFICATION COST AND SCHEDULE

R. O. M. cost estimate: $162K

Duration: 10 months

These are preliminary engineering estimates based on results for sensor modification of this type as developed by General Electric in a study for NASA/LARC titled "Shuttle Experiment Integration Study." Included in the cost estimate are the modification analysis and design, rework and refurbishment, and purchase of new components required for adaptation to EVAL conditions. The instrument is currently undergoing tests at Ball Brother Research Corporation.

4.3 ECLECTIC SATELLITE PYRHELIOMETER (ESP)

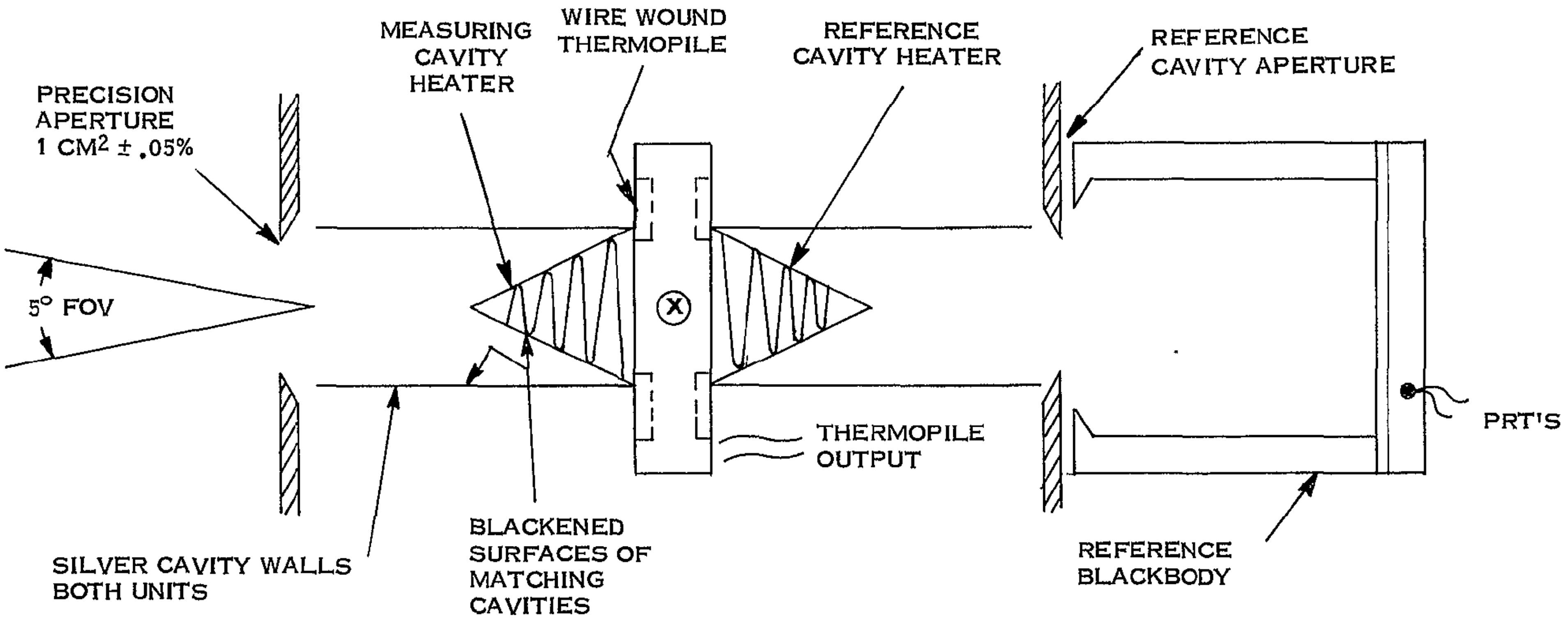
ESP consists of three radiometers utilizing self-calibrating cavity detectors and a set of fourteen filters applied to two of these radiometers. The instrument, which is designed to fly on the Solar Max and AEM missions, measures the solar constant of radiation in selected spectral regions, and monitors the variation of solar radiation to an accuracy and precision equal to or less than 1% and 0.2%, respectively. The instrument occupies an envelope of 0.2 x 0.3 x 0.3 meters and weighs approximately 11 Kg. A sketch of the ESP components is provided in Figure 4-3.

4.3.1 MODIFICATIONS

The sensor requires very few adaptations for Spacelab flight since it is compatible with spacecraft missions utilizing the Space Shuttle transportation system. A review of its capabilities showed the following modifications or interfaces to be necessary.

Structural

The sensor mounts will consist of de-tuned attenuators to reduce the mechanically-induced vibration during launch to acceptable levels. Although no acoustic protection of the entire assembly is deemed necessary, it is recommended that the effect of acoustically induced random vibration be analyzed with respect to critical radiometer components such as the cavity detectors. This will determine whether or not individual components that are susceptible to damage or changes in performance characteristics require dampening against noise.



A BALANCED WIRE WOUND THERMOPILE TYPE WITH MATCHED HEATABLE CAVITY RECEIVERS, MATCHED PRECISION APERTURES AND AN IN-FLIGHT BLACKBODY REFERENCE SOURCE - CAVITY FUNCTION IS REVERSED BY ROTATION ABOUT X

Figure 4-3. Sketch of Basic Components of ESP Detector

Optical

No modifications are required.

Thermal

Insulation blankets are required to protect the instrument against solar flux during portions of the orbit when the sun will illuminate the sensor.

Electrical

The sensor will accept the raw power input of 28 VDC from the Spacelab and will perform the power conditioning functions internally within the instrument. No power conditioning circuitry will be required.

A new harness will be required for compatibility with the Spacelab Electrical Power Subsystem.

Command and Control

Provisions will be required for execution of the command functions for the gimbal system described in a latter paragraph. A gimbal control servo system will be required for azimuth and elevation pointing and tracking.

The instrument electronics must be modified to accept the command signal levels and format of the Spacelab C&DMS.

Data Management

The data output signal level and format should be changed to make it compatible with the EVAL/Spacelab data handling system; this will require some modifications to the present instrument electronics.

Contamination

No requirement.

Pointing and Stabilization

A two-axis gimbal will be required to permit the instrument to point to the sun within $\pm 1^{\circ}$. The angular excursion and slewing rate specifications will be dependent upon the orbit, mounting position, and attitude on the standard test rack.

4.3.2 MODIFICATION COST AND SCHEDULE

R.O.M. cost estimate:	$222K
Duration:	11 months

These estimates are based on an engineering analysis performed by GE, and include the modification analysis and design, refurbishment, tests, and integration of the sensor. The instrument is presently located at NASA/LARC.

4.4 SCANNING MULTICHANNEL MICROWAVE RADIOMETER (SMMR)

SMMR is a passive microwave instrument which measures microwave thermal emission from the earth's atmosphere and surface in five channels: 6.6, 10.69, 18.0, 22.05 and 37 GHz. Each channel measures radiation in two orthogonal linear polarizations. A mechanically scanned parabolic antenna focuses microwave thermal emission into a multi-frequency feed system which is connected to five separate radiometers through orthomode transducers. The sensor, which weighs approximately 46.7 Kg, employs an antenna aperture of 80 cm, resulting in an overall envelope of 80 x 80 x 15 cm. Figure 4-4 illustrates this sensor.

The brightness temperature data will be used to estimate surface temperature profiles, water vapor densities, storm cell structure, sea surface temperature, sea surface wind velocity, snow cover, soil moisture and ice movements.

4.4.1 MODIFICATIONS

This instrument, designed for flight on automated satellites such as Nimbus G and Seasat, will require few modifications. The most significant modifications are those required due to the lower orbital altitude of the EVAL missions as compared to the satellite missions. The following is a summary of the required changes.

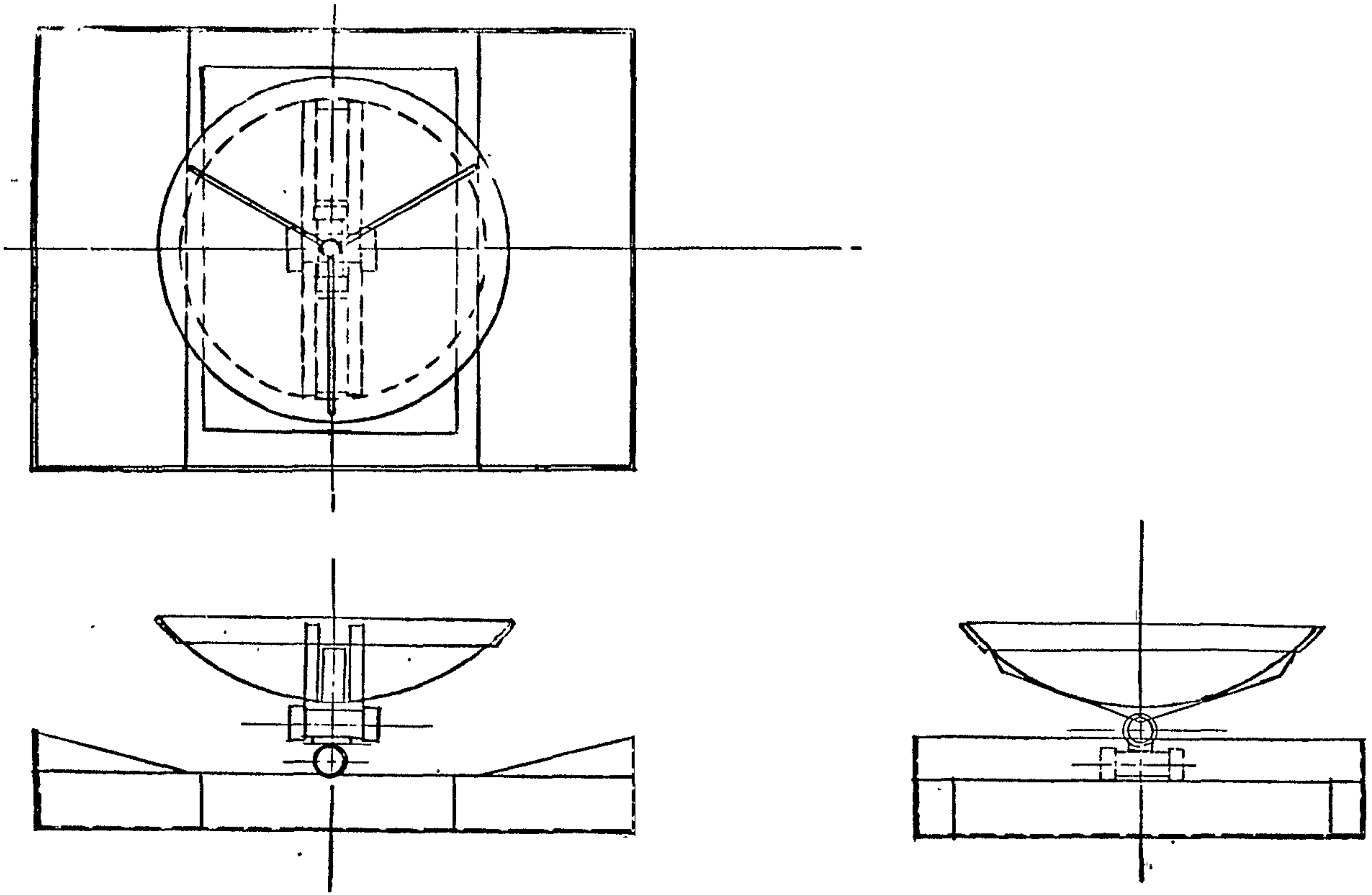


Figure 4-4. Scanning Multichannel Microwave Radiometer

Structural

The parabolic antenna structure should be analyzed to determine its susceptibility to damage or deformation due to acoustically induced random vibration. Reinforcement of the antenna is likely, as well as the addition of acoustic dampener material to the non-reflecting surface to attenuate the lower frequencies.

A redesign of the antenna drive caging mechanism will be required - primarily to make provisions for recaging the drive during Shuttle reentry, but also to eliminate the use of pyrotechnics for uncaging after orbital placement.

Due to the lower altitude of the EVAL mission, adjustments will be necessary in the antenna swath angle and beamwidth in order to ensure proper ground coverage and swath overlap. These adjustments will result in design changes in the feed horn and parabolic antenna. It is anticipated that the feed horn and antenna on the present unit will need to be replaced with new ones, since the changes will be substantial.

Thermal

A thermal analysis will be required to determine the thermal protection for the instrument under the conditions of the Shuttle environment. The design goal will be to use a cold plate at the instrument mounting interface to permit conduction of the 56 watts of heat dissipation. Connection between the cold plate and the Spacelab cooling-water loop can be made through the Experiment Heat Exchanger.

The scan drive assembly may be enclosed in multilayer insulation supported by a silica cloth cover. The reflector front surface may be protected by means of a metallized coating, and the reflector back surface and support structure can use a suitable thermal coating such as aluminum silicon paint.

Electrical

The present SMMR unit for Nimbus G is designed to accept negative polarity 24.5 VDC regulated power; whereas the Shuttle power bus potential is +28 VDC. A suitable polarity converter will be required. This may be a copy of the converter which is being built for the Seasat SMMR.

Command/Control

The scan drive mechanism will require modification to permit a wider scan angle (estimated at $\pm$ 35 degrees). The scan speed will be adjusted to conform with the mission orbital parameters and swath width.

The command circuitry for the instrument must be modified to render it compatible with the EVAL/Spacelab data handling system's signal level and format. The SMMR command functions require - 23.5 VDC driver (-7 VDC min.)@200 milliamps, for 50 $\pm$ 15 milliseconds; the Spacelab Remote Acquisition Unit produces on/off command voltage level of +5 $\pm$ 1.0 VDC, 20 milliamps, for 30 milliseconds.

One of the command subsystem options in EVAL if adopted may make it unnecessary to modify the SMMR command electronics: The commands would be routed through the Spacelab computer to the Remote Acquisition Unit (RAU) to a circuit provided by EVAL consisting of a command decoder, multiplexer, output register and buffer amplifiers. The latter would be designed to drive the various EVAL instruments (including SMMR) at their required signal characteristics.

Data Management

The signal conditioning circuitry in the instrument must be modified to be compatible with the EVAL/Spacelab Command and Data Management Subsystem C&DMS. The SMMR data output signal and format are compatible with the Nimbus "VIP" system, which requires different signal loads (e.g., 5 $\pm$ 0.8 VDC vs. 3.5 $\pm$ 1 VDC) and pulse width/synchronization than those in the Spacelab C&DMS.

Electromagnetic Interference

The instrument will be susceptible to electromagnetic interference within the following frequency bands:

RF Channels:	6.450	to	6.750	GHz
	10.540	to	10.840	GHz
	17.850	to	18.150	GHz
	21.900	to	22.200	GHz
	36.850	to	37.150	GHz
IF Channel:	10	to	110	GHz

Although specific sources of interference within these bands have not been identified, it is anticipated that countermeasures will have to be built into the instrument electronics to prevent such interference from the Shuttle avionics system and other payloads on board.

Pointing and Stabilization

The instrument will not require a gimbal system; however, an attitude reference system will be required at the sensor platform (e.g., pallet or standard test rack) to provide position knowledge of the platform to within approximately $\pm$ 0.5 degree.

4.4.2 MODIFICATION COST AND SCHEDULE

R.O.M. cost estimate:	$367K
Duration:	14 months

This is a preliminary engineering estimate based on general cost data for sensor modifications of this type generated by GE. Included in the cost estimate are the design, fabrication, test, and integration activities necessary to make this sensor compatible with an EVAL flight. The instrument development has been sponsored by NASA/GSFC, and is presently located at GE, Valley Forge, Penna.

4.5 MONITORING OF AIR POLLUTION FROM SATELLITES (MAPS)

The instrument measures concentrations of CO, CO_2, SO_2, NO, NO_2, NH_3, and CH_4 in the range of 0.001 ppm to 350 ppm. Optical correlation of gases through a gas filter correla-

tion analyzer permits selective measurement of the change in infared radiation in the 2 to 20 micron range due to specific pollutants. The measurement of tropospheric pollutants will permit the determination of constituent dispersal rates and longterm buildup to forecast regional pollution and establish relationships with global meteorology. Chemical processes and sink mechanisms in the upper atmosphere will also be investigated through MAPS measurements.

Overall dimensions of the instrument are 32 x 32 x 20 cm; the weight is 125 Kg. Figure 4-5 indicates the MAPS configuration.

4.5.1 MODIFICATIONS

Structural

The optical head assembly, consisting of the gas modules, pre-amps, and calibration sources may be susceptible to damage due to the Shuttle dynamic environment during boost. The recommended approach is to inspect the AAFE configuration of the instrument and effect re-inforcement and acoustic dampening of critical components such as the gas modules. A vibration and acoustic test is recommended for the instrument assembly. Gas leakage tests must be performed on the gas cells, to insure the integrity of the current cell configuration.

Optical

No optical modifications are required.

Thermal

The AAFE instrument has a double jacket insulation and heat dissipating air flow chamber for operation during hot pre-flight ambient conditions in aircraft. This insulation jacket must be replaced with suitable space type insulation. A cold plate will be required to provide a sink for the electronic assembly. The cold plate would be connected to the Spacelab cooling water loop through the Experiment Heat Exchanger.

The environmental control of the optical head must be redesigned to be compatible with the Shuttle mission conditions. The thermoelectric cooler heat sink must be modified to dump 30 watts into a stable heat sink at $25^{\circ} \pm 1^{\circ}C$.

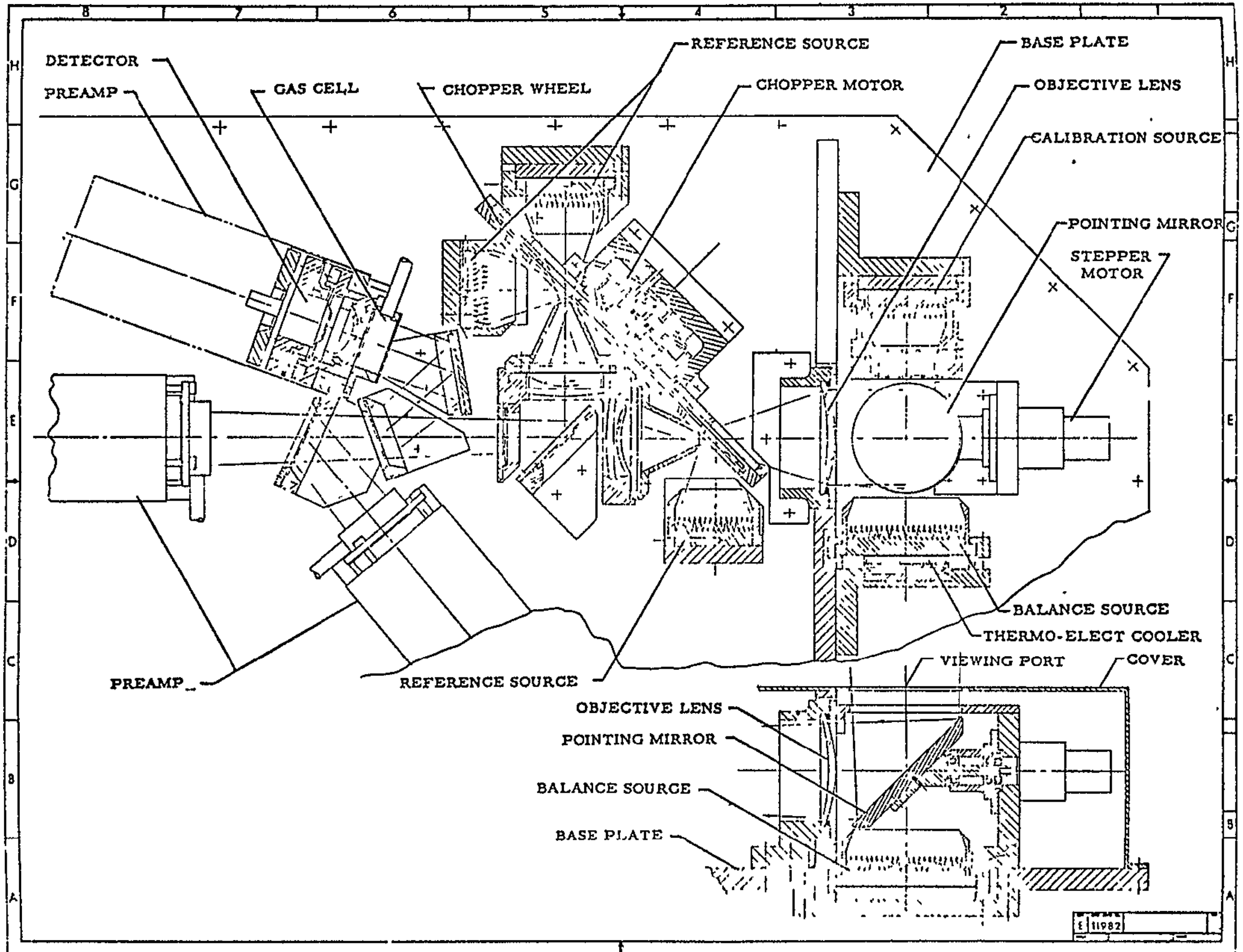


Figure 4-5. MAPS Electric-Optical Head Layout

Electrical

Power requirements are 150 watts at 28 VDC and 50 watts at 110 V 400 Hz. The AC power requirement will necessitate the use of a 400 Hz experiment inverter.

Command/Control

New command circuitry will be required to interface the instrument controls with the EVAL/ Spacelab Command and Data Management Subsystem. A total of 26 commands will be required.

Data Management

The signal conditioning circuitry in the instrument should be modified to be compatible with the EVAL/Spacelab data system. The instrument data rate will be approximately 1 Kbps.

Contamination

Provisions will be required for sealing the optics during launch, ascent, and the initial portion of the orbital mission to prevent gaseous and particulate deposition on the optics.

Pointing and Stabilization

No special requirements or modifications are anticipated.

4.5.2 MODIFICATION COST AND SCHEDULE

R.O.M. cost estimate:	$280 K
Duration:	10 months

The cost and schedule estimates for the required modifications have been generated by GE based on cost information data obtained from the "Shuttle Experiment Integration Study." The design, fabrication, test, and assembly of the sensor are included in the estimates. NASA/ LARC is the custodian of this sensor.

4.6 LASER RANGING SYSTEM (LRS)

The LRS instrument is essentially a short pulse ($\sim 10^{-9}$ seconds) Nd: YAG laser transmitter which is integrated with optical detectors and electronics such that the transmittal time of the pulse to a ground target and back can be measured. The system is being designed for targeting on ground-based corner-cube retroflectors to monitor small scale motions

of reference points on the Earth's surface. However, with minor modifications the system can be utilized to perform cloud physics investigations which will be complementary to the measurements carried out by the Cloud Physics Radiometer.

The instrument will have a weight of approximately 60 Kg and will occupy a volume of about 0.25 m^3. A sketch of the instrument is given in Figure 4-6. These values are exclusive of any pointing system which may be required.

4.6.1 MODIFICATIONS

The LRS is being designed as a Spacelab experiment, consequently interface problems will be considered in the routine development of the system. This instrument presently exists as a breadboard model. The comments below illustrate the type of interface which will be required with the Spacelab facility.

Structural

Most of the components required for the LRS exist in a ruggedized state as a result of development for past projects. However, they must be assembled into an integrated package which will survive the Shuttle launch environment.

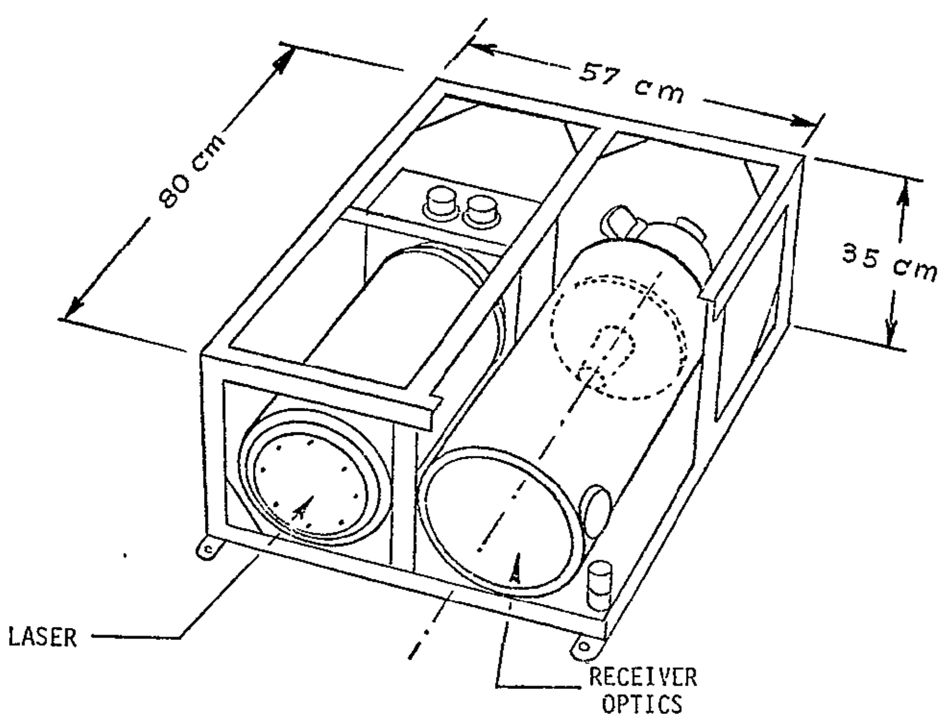


Figure 4-6. Laser Ranging System (Signal Processor and Power Supplies not Shown)

Optical

The optical system must be designed for compatibility with the with the Cloud Physics Radiometer. Consideration must be given to the type of scan capability desired for cloud physics investigations which will not conflict with the ability to make precise range measurements on ground targets.

Thermal

It is expected that the 200-300 watts power dissipation required by the LRS can be achieved with proper interfacing to the cold plate available on Shuttle.

Electrical

Power required to operate the LRS is approximately 250 watts average (not including power for the pointing system). Harnesses must be designed to interface between the pallet-mounted laser/receiver and the module-mounted instrumentation (e. g., mini-computer, tape recorder, etc.).

Command/Control

The LRS will be turned on/off manually with intermediate operation controlled automatically via a mini-computer.

Data Management

The total quantity of data accumulated by the LRS will depend on its mode of operation, i. e., targeting on ground-based retroreflectors or targeting on clouds. The latter mode will result in a larger volume of data but in either event the High Rate Data Multiplexer will not be required. It is expected that the analog wave-form data will be A/D converted and stored on magnetic tape and dumped periodically as required.

Contamination

The LRS instrument does not generate any type of contaminant, but its operation would be adversely affected by any effluents which contaminated the optical system. The receiver optical system should be hermetically sealed in order to minimize such contamination.

Pointing and Stabilization

The pointing and stabilization required by the LRS represents the only major problem to be solved. Operating as a ranger, to ground-based retroreflectors, a pointing accuracy of $\pm$ 0.5 mr for 10-15 sec is required. A potential solution to this problem is to utilize the Small Instrument Pointing System (SIPS). When the LRS is used for cloud physics investigations the pointing requirements are not as severe and the accuracy provided by the Shuttle itself may be adequate.

4.6.2 COST AND SCHEDULE

The technology required to develop the LRS into a flight-ready instrument for a 1981 Shuttle launch exists at the present time. The estimate cost required to develop, fabricate and test the instrument (exclusive of the pointing system) is approximately 2 million dollars, and would take 18 months.

4.7 CLOUD PHYSICS RADIOMETER (CPR)

The CPR is an eight channel scanning radiometer with seven channels in the near infrared and one channel in the thermal infrared at 11μm. The instrument will be used to determine a minimum of six physical properties of optically thick clouds including cloud top altitude, thermodynamic phase, particle density, size and temperature, optical thickness, and possibly water vapor mixing ratio.

The instrument will have a weight of approximately 190 kg and occupy a volume of approximately 0.1m^3. Figure 4-7 provides a pictorial of the sensor.

4.7.1 MODIFICATION

An aircraft version of the CPR exists and has recently undergone flight tests. In order to insure compatibility with the EVAL on the Shuttle, a few minor modifications, as described below, will be required.

Structural

Vibration tests will be required on the existing aircraft version of the CPR to determine what modifications, if any, are required to insure survival of the Shuttle launch environment.

Optical

It is not expected that any modifications in the optical system will be required other than a change in the mirror scan rate to compensate for the different v/h of the Shuttle.

Thermal

The aircraft version of the CPR utilizes two LN_2 dewars for cooling of the detectors and bandpass filters. In order to extend the operating time of the instrument to a seven day mission, a closed cycle cooling system will be required.

Electrical

The electronics must be repackaged for Spacelab flight. Command and data transmission lines can be provided to interface between the pallet-mounted detector and the module-mounted electronics. A hazard warning capability should also be included.

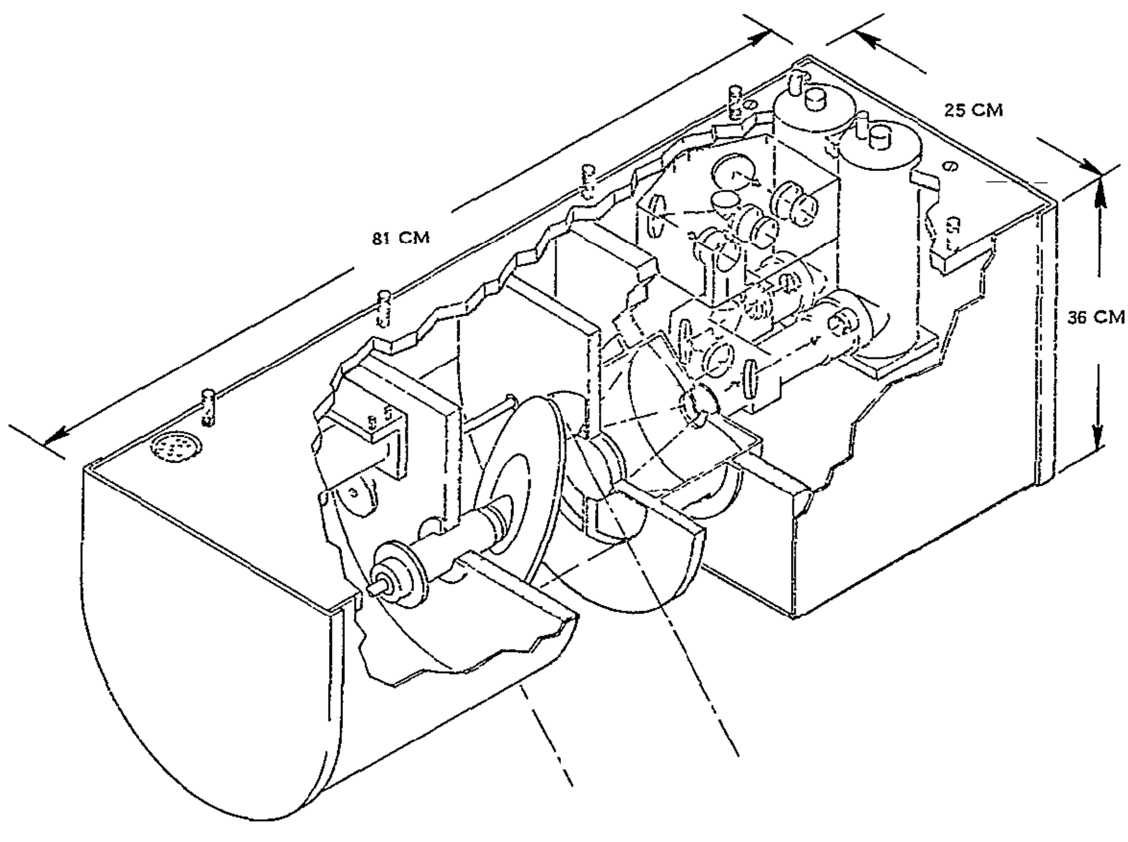


Figure 4-7. CPR Optical System

Command and Control

Commands to uncover the optics and turn the CPR on will be supplied manually by the payload specialist as targets of opportunity arise. After turn-on, the instrument operation is automatic until it is turned off manually or by a preset automatic timer.

Data Management

The existing data system must be repackaged to be compatible with the Spacelab data handling system. It is expected that the A/D conversion will be performed by the CPR electronics. The possibility of monitoring the instrument status through the RAU interface should be considered.

Contamination

The CPR will be self-protecting from contaminants during launch operations. However, any effluents in the vicinity of the optical system during in-flight operation would reduce the quality of the data obtained.

Pointing and Stabilization

The pointing accuracy provided by the Shuttle, itself, will be adequate.

4.7.2 COST AND SCHEDULE

The estimated cost required to provide a flight-ready instrument for a Spacelab flight is approximately $200K. Included in this cost estimate are the modification analysis and design, rework, and refurbishment required for adaptation to EVAL. It would take 12 months to procure it. The instrument is presently being proposed for Space Shuttle OFT-2 missions, and is currently at NASA/GSFC.

4.8 RADIOMETER/SCATTEROMETER (S-193)

The S-193 is a 13.9 GHz sensor developed to operate as a passive radiometer (RAD), active scatterometer (SCAT), and altimeter on Skylab. For EVAL, only the RAD and SCAT portions are being considered. The mechanically scanned 44.5"D parabolic dish antenna is two-axis gimbal mounted on the sensor electronics box which is about 213 cm long, 53 cm wide and 18 cm high. The RADSCAT antenna can scan in-track forward 52^{o} and cross-track scan left or right 52^{o}. The S-193 weighs about 140 kg as it is shown in Figure 4-8.

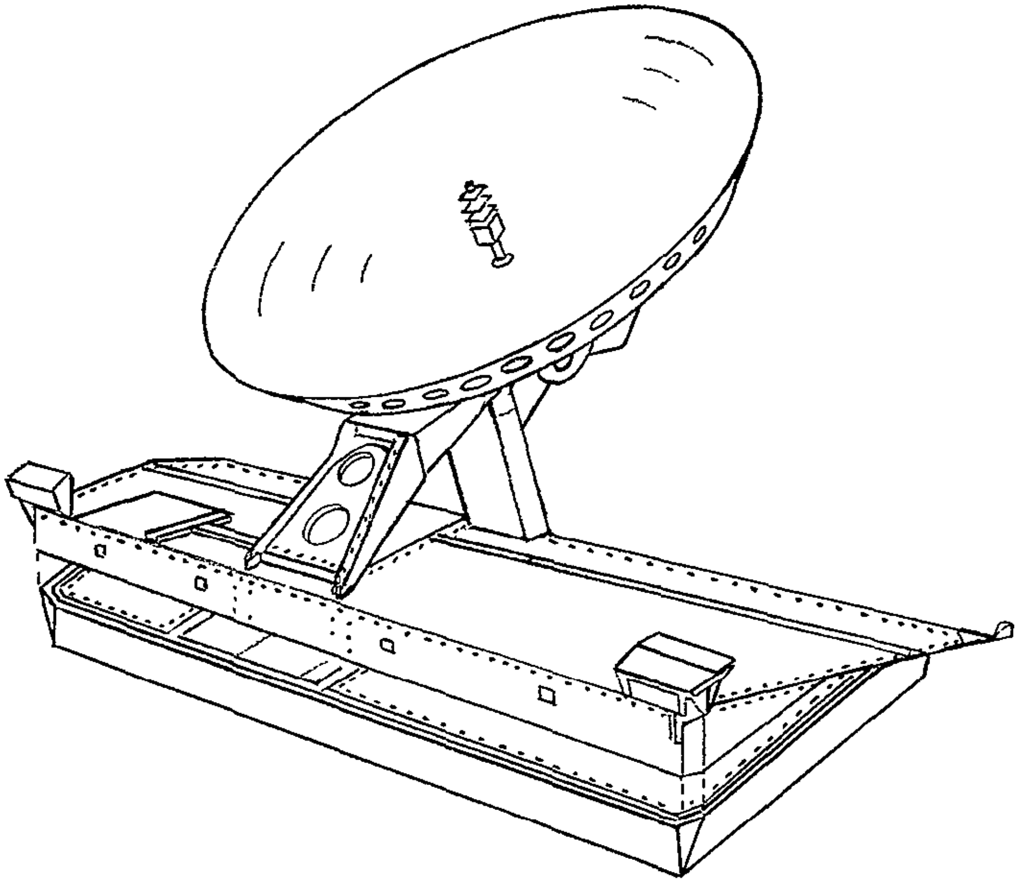


Figure 4-8. S-193 Radiometer/Scatterometer

The RADSCAT measures microwave thermal radiation emission and radar scattering coefficient from oceanic and terrestrial surfaces.

4.8.1 MODIFICATIONS

The S-193 as developed will require the following modifications and adaptations to be compatible with the EVAL facility on Spacelab.

Structural

A simple angled adapter support will be required to cant the sensor electronics box at about 40° from the pallet floor. The support should have sufficient height to permit the antenna to scan its full range without obscuration of the antenna beam.

Thermal

Preliminary examination of the thermal environment indicates that no redesign is required.

Electrical

The S-193 operates from a 24-30 VDC bus. The overall average power requirement is 206 watts with a peak during antenna slew of 291 watts. No modifications are required.

Command/Control

The S-193 functions from the EREP (Earth Resources Experiment Package) Command and Display Panel operated by an astronaut. All command/control logic circuits in the S-193 are expected to be fully compatible with the Spacelab Computer.

Because of the lower altitude of Spacelab, the SCAT pulse length and pulse repetition frequency will have to be modified for optimum operation. The RAD integration time will have to be modified but the effect will have minimal system impact.

Data Management

The RADSCAT PCM output bit rate is 5.33 kBPS. Status signals in both digital and analog formats are normally sent to the EREP C&C Panel. For autonomous, computer controlled operation, the status signals should be commutated into a status data stream.

Pointing and Stabilization

The S-193 requires a platform stability of $\pm 0.5^{\circ}$ in all three axes which is compatible with Spacelab.

Electromagnetic Compatibility

Within the S-193 the extremely sensitive radiometer is fully protected from any interference caused by the travelling wave tube transmitter of the scatterometer.

4.8.2 COST AND SCHEDULE

Modification of the SCAT pulse modulator is estimated to cost about $50K and take about 6 months.

The existing S-193 was a backup unit instrument for the Skylab ERAP package. The instrument has been separated into the Radiometer-Scatterometer (presently at NASA/GSFC) and the altimeter (presently at NASA/Wallops). The cost specified came from engineering estimates provided by GE's design section - who developed the original instrument.

4.9 GEOS-C ALTIMETER (ALT)

The GEOS-C Altimeter operates at 13.9 GHz and measures the height above the earth's surface. The nadir oriented dish antenna is about 2 feet in diameter. The outer dimensions of the sensor package are 27.4"D cylinder and 28.4" high. The weight is about 150 lbs. This sensor was designed for the GEOS satellite.

4.9.1 MODIFICATIONS

The GEOS-C will require the following modifications and adaptations to be compatible with the EVAL facility on Spacelab.

Structural

A simple adapter ring support is required to fasten the GEOS-C altimeter to the pallet. A clear view by the antenna beam is required.

Thermal

Preliminary examination of the thermal environment indicates that no redesign is required.

Electrical

The GEOS-C operates from a 28 VDC bus. The power requirement is 150 watts (no change in power level for EVAL).

Command/Control

The GEOS-C altimeter is designed to operate autonomously by either stored or direct command by a computer.

Because of the lower altitude of Spacelab, the receive blanking gate will have to be shortened from the current 3.2 msec to about 1.5 msec.

The GEOS-C sensor requires a 5 MHz reference clock signal from Spacelab.

Data Management

The output data rate from GEOS-C is 15.6 kBPS when operating in the intensive mode. (No change in data rate for EVAL.)

Pointing and Stabilization

The GEOS-C altimeter requires a platform pitch and roll stability of 0.5° from nadir.

Electromagnetic Compatibility

Within the GEOS-C altimeter its sensitive receiver is fully protected from its high peak power transmitter output pulses.

4.9.2 COST AND SCHEDULE

Modification of the receiver is estimated to cost \$100K and take about 6 months. The life condition of the traveling wave tubes will have to be assessed for orbit operation.

4.10 SOLAR BACKSCATTER UV & TOTAL OZONE MAPPING SPECTROMETER (SBUV/TOMS)

The SBUV/TOMS provides both total synoptic and sampled vertical ozone distributions to an altitude of 60 Km. The UV spectrometer measures solar UV that is back-scattered by the earth's atmosphere at 12 wavelengths between 2500°A and 3400°A with a spectral bandpass of 10°A.

The ozone mapper, operated in parallel with the UV spectrometer, has a step scan ± 0.89 rad, normal to the orbital track with an IFOV of 0.052 rad. At each scan position the earth's radiance is monitored at four wavelengths between 3100 and 3400 A°, and at 3800 A° to infer the total ozone amount. The sensor weight is 15.5 kg., and is configured as shown in Figure 4-9.

4.10.1 MODIFICATIONS

The SBUV/TOMS is planned to be flown on the Nimbus G Spacecraft and will require minor modification to be flown on the Shuttle.

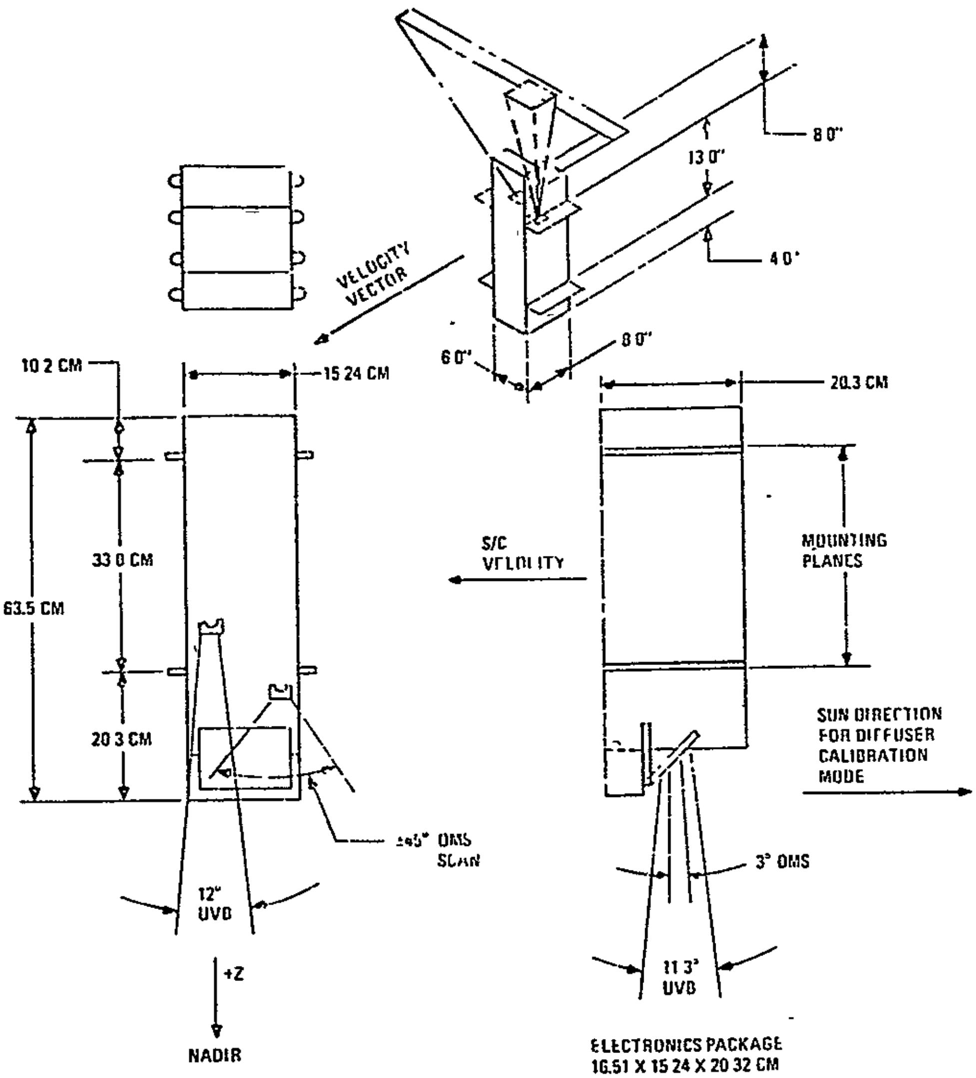


Figure 4-9. SBUV/TOMS Configuration

Structural

Some modification is required to provide external attachment points compatible with the Shuttle pallet or SEOPS.

Thermal

No thermal redesign is required. However, in the SEOPS concept, thermal insulation will be required from the SEOPS structure by using thermal isolators.

Electrical

The instrument operates from a 28 VDC bus. The overall average power requirement is 10 watts (no change for EVAL).

Command and Control

The command and telemetry interface can be compatible with Spacelab/SEOPS by using the remote MDM interface capability of the orbiter to send command and receive telemetry.

Data Management

The majority of modifications required to use this instrument on Shuttle in a low earth orbit (<300 n. m.) are in the ground data processing due to the improved resolution, the wide range of sun angles, and the changes in coverage geometry. On EVAL, the major change is the interface circuit required for storage of the 600 BPS data on the EVAL narrowband tape recorder as well as the necessary ancillary data for processing.

Since the SBUV/TOMS is a sampling instrument, it does not require a great deal of modularization to accommodate altitude and inclination changes. Corrections for sample size and sun angle can be performed in the ground processing. The capability exists, however, to modify the sample wavelengths and spectral resolution as a function of specific mission requirements.

Pointing & Stabilization

The Shuttle is adequate and no modification is required.

4.10.2 COST AND SCHEDULE

To modify and procure this instrument would cost $500K and take 18 months.

This estimate was provided by the manufacturer (Beckman) and includes the fabrication, test, and integration of a new sensor - at present only one unit is being built, and it is scheduled to be flown in Nimbus G.

4.11 LARGE FORMAT CAMERA (LFC)

The large format, wide angle camera is a modification of a design investigated by Itek to satisfy a need for a very precise metric camera. This camera can be used for precise applications such as needed by the US Coast & Geodetic Survey. The Shuttle version of

this 9 x 18 inch (format) camera will basically be the same lens, shutter, and film transport system as the precision metric camera used on aircraft flights.

The large Format Camera (LFC) comes with a 12", 18" & 24" focal length lenses. The frame size on the ground is 38^{o} cross-track and 74^{o} in-track

4.11.1 MODIFICATIONS

The framing camera will require the following modifications to the current design to be compatible with the EVAL facility:

Structural

The mechanical configuration for the LFC with 12 inch focal length is shown in Figure 4-10. For EVAL a supply of nitrogen capable of delivering up to 4 x 10^{3} lbs/min is required for pressurizing the film transport enclosure in order to avoid corona discharge. The platen also has to be modified for EVAL.

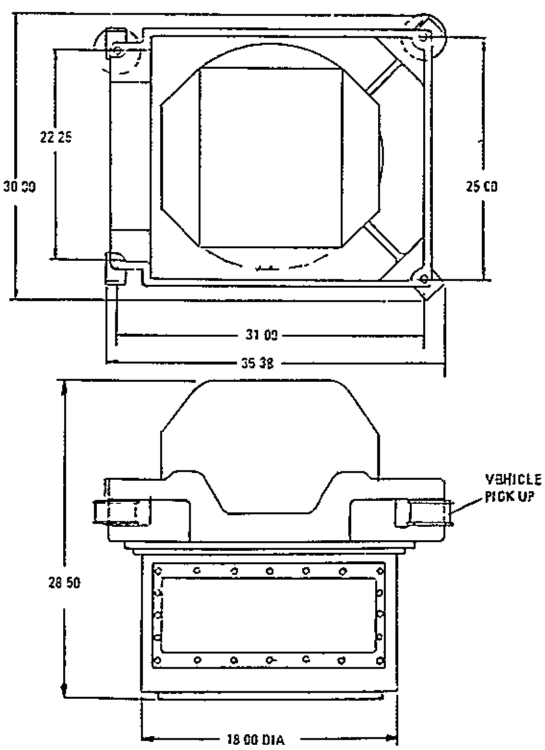


Figure 4-10. Large Format Camera Outline

Thermal

The LFC requires thermal control to a preset temperature as well as thermal stability sufficient to maintain thermal gradients. The camera includes heaters and controllers to maintain it at a temperature of $70^{\circ} \pm 1^{\circ}F$. Multi-reflective layer insulation must be used to isolate the camera from adjacent heat sinks and sources. New mounting brackets will have to be designed to minimize heat conduction to the EVAL structure as well as to insure maintenance of pre-launch alignment and prevent external stresses from distorting the camera alignment.

A thermal door must be placed in front of the entrance aperture to reduce the heat loss during non-operating periods. This door will also be used to prevent the condensation of contaminants on the optical surface.

Electrical

The camera requires 60 watts continuous for thermal control. During operation, an additional average power of 120 watts with pulse peaks of 700 watts is required during the exposure time of 10 to 20 ms.

The camera can operate from a 28 VDC $\pm$ 4V power bus and does not require any modification. Motors, brushes, and tachometers will have to be replaced with space qualified hardware.

Command & Control

The LFC requires time reference information during each exposure and can be operated fully automatically. It also requires calibration by a starfield calibrator both prior to launch and after return.

About 70 channels are required for telemetry, with an additional 75 channels for diagnostic information for automatic performance checkout. These are modifications from the present design.

Data Management

All information is stored on film. A data block has to be added to the format.

Pointing & Stabilization

The Shuttle Orbiter attitude stability is specified as $\pm$ 0.001 deg/sec and the pointing accuracy of 0.5 degrees with a possibility of about 2.0 degrees due to the Shuttle thermal gradients. For the range of altitudes between 120 to 400 mm and the specified lenses, no modifications have to be performed to the present design.

4.11.2 COST & SCHEDULE

A modified new camera would cost approximately $5M and take 36 months to procure. This is an engineering estimate obtained from ITEK to build and space-qualify a new version of their aircraft camera.

4.12 OBJECT PLANE LINEAR SCANNER (THEMATIC MAPPER-TM)

This instrument is a mechanical scanner proposed for Landsat-D. The object plane scanning is obtained by directing the ground scene with an oscillating scanner mirror through a telescope and relay optics to a series of detectors located at the focal plane. Spectral definition is obtained by a series of band pass filters, with spectral separation into the seven spectral bands obtained by spatial separation. Data is taken on each half cycle of the scanning mirror oscillation by use of an image-motion-compensation dual mirror arrangement located in the optical system. The instrument weighs 180 kg and measures 112 x 93 x 35 cm. Figure 4-11 portrays this sensor.

4.12.1 MODIFICATIONS

The object plane linear scanner developed by Hughes will require the following modifications to the existing design:

Structural

The major scanner design modification required to the Landsat-D design concept in order to make the scanner compatible with EVAL mission objectives is the incorporation of a variable frequency scan mirror drive. With fixed scan angle and zero scan overlap, the scan mirror frequency increases as orbital altitude decreases; therefore a variable control is required.

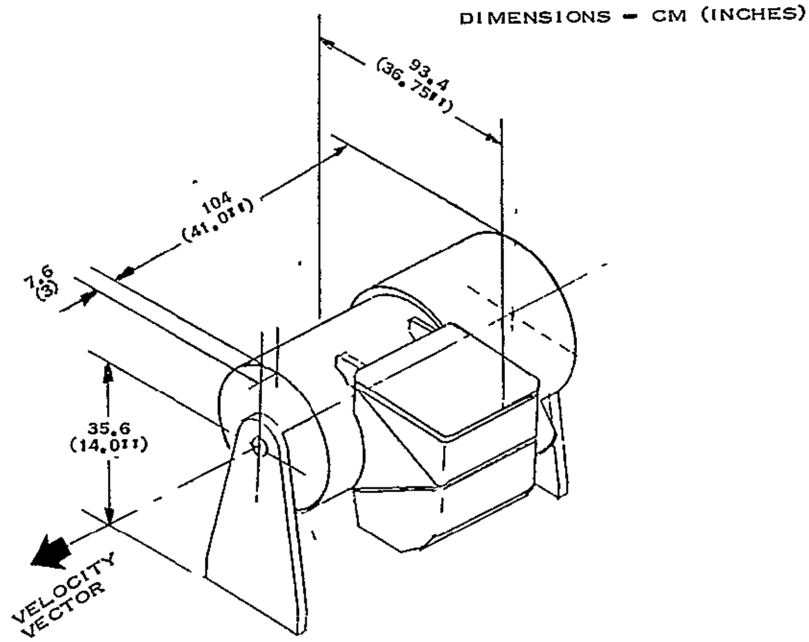


Figure 4-11. Object Plane Linear Scanner

The increase in scan mirror frequency does not appear to present any mechanical problems down to an operational altitude of approximately 200 nm; however EVAL flights are frequently flown at altitudes as low as 120 km.

For altitudes below 200 nautical miles it would be possible to modify the focal plane configuration to increase the IFOV per detector; thereby decreasing the scan frequency requirement. For example, in order to maintain a maximum scan mirror frequency of 18 Hz at a 100 nautical mile altitude, the IFOV would need to be increased to approximately 70 m rad as compared to the nominal 35 m rad. This would still provide high ground resolution. This change is included in the proposed sensor modifications.

Thermal

Replace the passive radiative cooler with a solid cryogen, Joule-Thomson or closed cycle cooler.

Electrical

The TM requires an average of 100 watts at 28 VDC - no modification is required in the power supply.

Command & Control

No information is available on the present command & telemetry requirements.

Data Management

Data rate will increase as the altitude. Thus, the electronics and interface with the tape recorder would need to be modified for the maximum data rate (120 Mbps) at the minimum altitude, and over-sampling will occur at higher altitudes.

Pointing & Stabilization

Off nadir pointing capability can be provided by rotating the whole instrument about its optical axis. This requires a tilt mechanism which is presently not part of the TM.

4.12.2 COST & SCHEDULE

A modified new TM would cost $ 8M to $ 10M and take 36 months to procure.

This is an engineering estimate generated by General Electric in conjunction with Hughes - one of the contractors involved in the development of this sensor.

4.13 CORRELATION INTERFEROMETRY FOR MEASUREMENT OF ATMOSPHERIC TRACE SPECIES (CIMATS)

The instrument is a two channel interferometer, one operating in the non-thermal infrared (2 to 2.4 μm) and the other in the thermal infrared (4 to 9 μm). A PbS detector operating at 195 ^{o}K is used in the 2 to 2.4 um channel and a HgCdTe detector operating at 77 ^{o}K in the 4 to 9 μm) channel. Each channel is capable of containing five (5) narrowband filters, thus providing the capability of making ten different spectral measurements. Two measurements, one in each channel, are made simultaneously with a measurement time of one second. This sensor has the capability of operating both in the nadir viewing mode and in the earth limb viewing mode.

Figure 4-12 is a schematic of the CIMATS sensor which is presently undergoing laboratory testing. The optomechanical portion of the sensor excluding the foreoptics (telescope) weights 27 kg and is 70 x 40 x 45 cm. Two interchangeable telescopes are available, a 7^{o} unit weighs 5 kg and is 40 cm long x 20 cm in diameter and a 2^{o} unit weighting 14 kg and 56 cm long x 36 cm in diameter. The electronics operate from 28 volts D. C., weigh 9 kg and are contained in a box 18 x 23 x 30 cm. Analog to digital conversion of the experimental data is accomplished in the electronics.

4.13.1 MODIFICATIONS

The current unit is designed for ground and aircraft environments, thus, several requirements for modification relate to survival and proper performance operation in the Shuttle boost and orbital environment. Other significant modifications relate to the instrument cooling and command and data management systems.

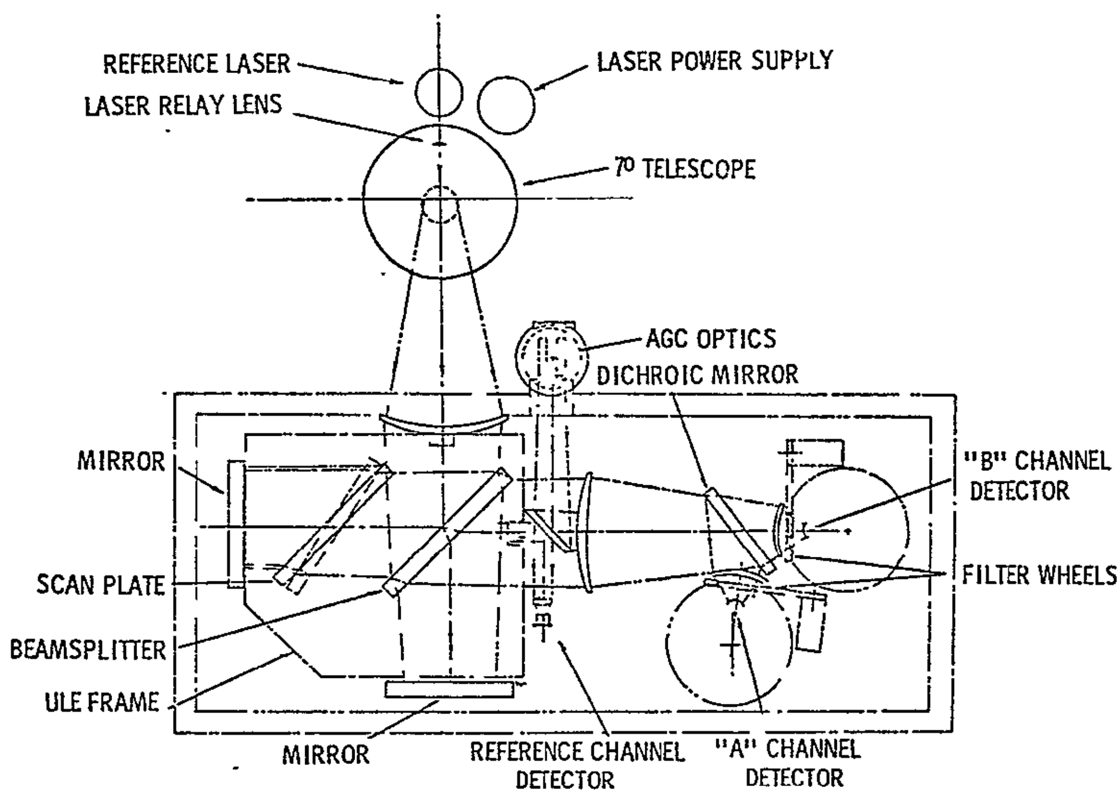


Figure 4-12. Correlation Interferometry for the Measurement of Atmospheric Species (CIMATS)

Structural

The acoustic noise level which the sensor will be exposed to should be reduced by a minimum of 10 dB. An enclosure of visco-elastic epoxy, as described for the LACATE sensor in section 4.1, can provide this reduction.

The scan plate/sensor arm may be subject to vibrational loads which could cause damage to the components. A vibrational analyses is required to define the extent of the problem and indicate the need for mechanical or electrical redesign to secure the components against the vibrational loads.

Three of the motors may require replacement. Although the scan plate torque motor drive is suitable for vacuum operation in its present configuration, a space qualified version of the same motor is available from the manufacturer and is preferred for this application. The two filter wheels in the optical paths to the detectors are driven by DC motors which would not be suitable for vacuum, however, versions of these motors suitable for a space environment are available.

The automatic gain control system contains a tuning fork chopper which may require replacement depending on the vibration environment. Replacement of the chopper with a new unit with stiffer blades will overcome the vibration problem. The electronics which drive, the chopper will have to be replaced with an encapsulated unit which will operate in vacuum. Since the new chopper may not have the same dimensions as the current unit, the housing which contains the chopper and blackbody may have to be redesigned and fabricated.

Optical

No optical modifications are needed due to the higher altitude of the EVAL mission over that of aircraft or balloon tests. The modifications to the optics for thermal reasons are described in section below..

Thermal

The sensor interior must be temperature controlled since the insulation and thermal control system were designed for field testing on the ground and in an aircraft. The insulation is not suitable for a spacecraft environment. As a result, a thermal analysis would be required to define the insulation requirements and verify the adequacy of the present thermal control system under the thermal loadings imposed by the Spacelab environment. The thermal analysis must also consider the heat dissipation from the power supplies which provide regulated power to the sensor and electronics. The power supplies are currently cooled by means of convectively cooled heat shields. In the Spacelab, heat dissipation by means of heat pipes or radiative cooling would have to be considered.

The reference blackbody is a temperature stabilized cavity type of blackbody which provides a known radiance source. The thermal insulation used in this component is a foam-in-place epoxy which would not be suitable for a spacecraft application. In the vacuum environment of space the foam insulation is not necessary to protect against conduction and convection cooling in the air space within the blackbody. A new blackbody of the same design fabricated without insulation or with multi-layer aluminized mylar insulation would be suitable.

The infrared laser which provides the 3.391 micron wavelength reference is not suitable for a vacuum environment due to outgassing of materials and high voltage in the power supply. The laser can be rebuilt to eliminate the outgassing and insulate the high voltage.

The present CIMATS detectors are mounted in dewars which require both dry ice and liquid nitrogen coolant. A mission study is necessary to define the measurement requirements with regard to species and accuracy. This will define the detector noise equipment power requirements, and hence the type of detector and coolant necessary. The current lead sulphide detector cooled at 196 ^{o}K can be replaced by a comparable thermoelectrically cooled unit. If the mission study requires the current mercury cadmium telluride detector to be cooled to liquid nitrogen temperature, then a liquid nitrogen transfer system would have to be designed or a detector system integrated with a closed cycle cooler system may be considered.

Electrical

The current sensor design uses many connectors and terminal blocks for electrical interconnections for testing purposes. The sensor will be rewired using hard-wire connections and filter qualified connectors. The sensor electronics will require repackaging and re-wiring to include the regulated power supplies and the power transistor for the sensor temperature control system, which are currently located in the sensor suitcase unit intended for monitoring and checkout functions.

Command and Control

The command circuitry must be modified to render it compatible with the EVAL/Spacelab data handling system's signal level and format.

Data Management

The signal conditioning circuitry in the instrument must be modified to make it compatible with the EVAL/Spacelab Command and Data Management Subsystem.

Contamination

A retractable lid is required at the optics aperture of the acoustic enclosure to maintain the optics sealed during launch, ascent, and the initial portion of the orbital mission. This is to prevent gaseous and particulate deposition on the instrument optics.

Pointing and Stabilization

An attitude reference sensing system will be required at the sensor mounting platform to provide accurate pointing during the limb measurement portion of the experiment.

4.13.2 MODIFICATION COST AND SCHEDULE

R.O.M. cost estimate:	$ 337
Duration:	14 months

This is a preliminary estimate based on inputs obtained from General Electric's Space Science Laboratory, where the instrument is under construction. Included in the estimates are the modification analysis and design, rework and refurbishment and new components required for adaptation to EVAL.

4.14 HIGH RESOLUTION IR SPECTROMETER (HIRS)

The Infrared Spectrometer is designed to obtain spatially independent IR radiances (that are unbiased with respect to cloud condition) at sufficient spectral and spatial resolutions so that the data may be used for determining the thermal structure of the earth's atmosphere. This instrument is a modification of the sensor currently in operation on Nimbus-F.

Basically, HIRS is a filter wheel device which scans normal to the orbit plane with a scan angle of $\pm$ 36.9° about the nadir for earth view. The optical telescope focuses the received radiant energy onto two cooled detectors and a photodiode which is used as a visible energy channel. Prior to reaching the detectors, the energy is spectrally separated into long wave (LW), short wave (SW), and a visible component, chopped and bandpass filtered. There are three detectors and 17 spectral bandpass filters. Figure 4-13 provides a sketch of this sensor.

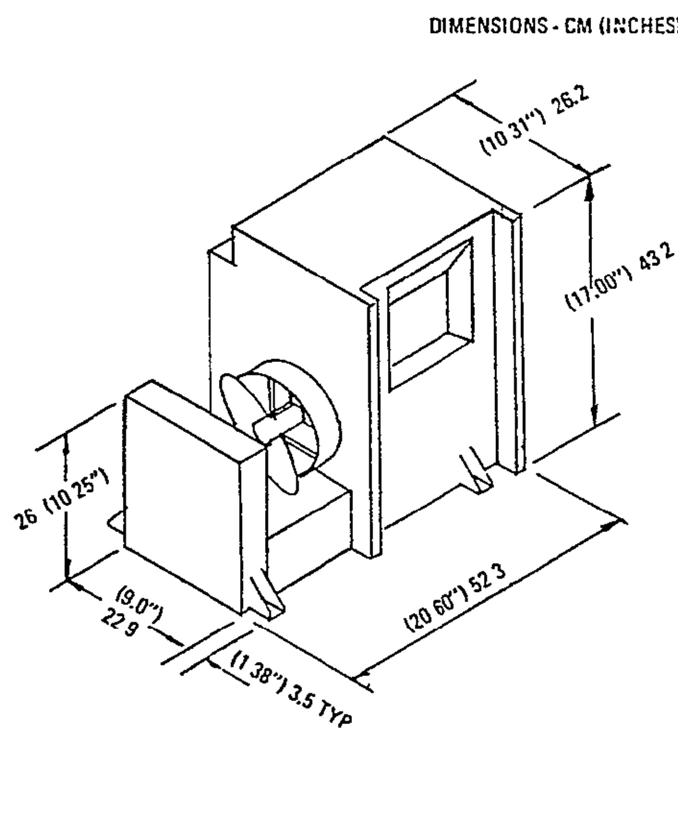


Figure 4-13. High Resolution Infrared Spectrometer

4.14.1 MODIFICATIONS

Only minor modifications are required for HIRS for Shuttle applications.

Structural/Thermal

The present HIRS cooler subassembly is a modular unit separable from the main frame. This subassembly contains two cooled detectors and their passive cooling structure. For EVAL applications this system must be replaced by an active cryogenic cooling system. A typical closed cycle system has been designed for similar applications of laboratory instruments for space use, and will be installed in the HIRS cooler module.

The sensor performance is also likely to be degraded by jitter. A method has been devised that would eliminate a gear set that drives one of the radiant energy choppers, resulting in an improvement in shortwave signal quality.

Reflective shields also are required as modifications to the system to prevent both earth and space input to the calibration target.

An improved method of radiant energy chopping has been devised that eliminates the need for a separate longwave chopper blade. This modification would eliminate a gear set that drives the longwave chopper and has caused degradation of system performance by its gear mesh noise (jitter). It will be most noticeable as an improvement in shortwave signal quality. This improvement is incorporated in the EVAL design version.

A 104.1° total view angle is required to permit viewing of space for a reference point. The space look is 65.7° from nadir. At an altitude of 200 nm the horizon is 70.9° from nadir, in which case the scan mirror would see the earth. Calibration would then be limited to the use of the two internal targets at 290° K and 255°K. The system could be modified to see space by removing the 255° K target.

A related problem is that of positioning the system on the Shuttle at a location that permits viewing space. If a space look is not convenient or possible, the internal targets are sufficient for system calibration. Reflective shields have to be added to the system to prevent earth and space vehicle heat input to the 255 K target.

Electrical

The EVAL payload will provide a positive input power to the Nimbus-F power source is -24.5 volts nominal, and supplies three inputs to the HIRS (F/C power, scan power, and electronics power). A switch in the polarity of the power being supplied to the censor is therefore required. Since all of the HIRS input circuits are isolated from the chassis, the switch to a positive +28 Vdc input would not be a problem.

Command & Control

Basically no modification is required for command & control.

Data Management

Basically no modification is required in this subsystem.

Pointing & Stabilization

The shuttle provides adequate attitude control support and no modification is required.

4.14.2 COST & SCHEDULE

There is a fully space qualified HIRS in storage available as a spare. To modify this instrument for Shuttle application would cost $200K and it would take 9 months.

This is an engineering estimate generated by the manufacture (ITT) who presently has the space instrument in storage.

4.15 INSTRUMENT MODIFICATION SUMMARY

A summary of the costs and schedules associated with implementing the modifications described in the preceeding pages is provided in Table 4-3. The major points to be gathered from this table are that 1) those sensors presently existing in a hardware state can usually be modified for a rather modest cost (several hundred thousand dollars) and within an elapsed time frame on the order of twelve months; 2) modification of sensor still in the design phase is a much larger undertaking. Typically, it is estimated that on the order of three years and several million dollars, dependent upon the sensor, will be required to modify the sensor for Shuttle flight, this results from the fact that not only must the sensor be redesigned, but it must also be fabricated, tested, and integrated.

Table 4-3. Instrument Modification Summary

Instrument	Modification Cost (K$)	Modification Schedule (Months)
LACATE	756	18
SAGE	162	10
ESP	222	11
SMMR	367	14
MAPS	280	10
LRS	2,000	18
CPR	200	12
S-193	50	6
ALT	100	6
SBUV/TOMS	500	18
LFC	5,000	36
TM	~ 9,000	36
CIMATS	337	14
HIRS	200	9

SECTION 5

CONCLUSIONS

The following conclusions are derived from the study results described in the preceding sections.

Dedicated Shuttle/Spacelab Payload

1. Earth viewing applications experiments/missions involving operational data gathering, technique development, sensor development, and end-to-end system demonstrations can be accomplished on Shuttle/Spacelab.

2. Significant synergistic benefits, both intra and cross discipline, can be derived by selective payload planning involving multiple experiments/missions

3. Cost effective payloads can be configured by commonizing on equipment and time-lining their operations

4. Items 2 and 3, above, are particularly evident when a particular payload is devoted to one or two disciplines.

5. All Spacelab module plus pallet configurations tend to exhibit undesirable longitudinal center of gravity locations.

6. Multiple passes should be planned over targets requiring visual observation since cloud cover can significantly reduce the probability of mission success (dependent upon the target area).

7. The Shuttle crew can efficiently be utilized to supplement the payload specialist(s) in payload operations

8. Very high data rates in excess of Shuttle/Spacelab capability will be a frequent payload characteristic, and will require special equipment for handling. In particular, some form of data compression is required for handling the data generated by the synthetic aperture radar. The thematic mapper also generates a high data stream which cannot be significantly compressed without losing data content. This necessitates the development of a very high data rate recorder.

9. Shuttle pointing and stability capabilities are inadequate for some experiments and must be supplemented by other systems.

10. There is relatively little power/energy available for sensor operation after the budget for Spacelab and other mission dependent/independent equipment is subtracted. The requirement for an additional energy kit will be a frequent occurrence. Also, the possible requirement for peaking batteries may exist dependent upon the final definition of Spacelab resources available to payloads.

11. The mission analyzed in this study was intended to explore the maximum stress limits of the system. However, practical consideration relating to weight, c.g., power, and thermal constraints indicate the undesirability of planning and flying missions requiring payloads of such magnitude and complexity.

Multi Mission Payload

1. The concept of a flexible, modular support structure for small payloads flying on Shuttle missions is feasible and practical.

2. A cradle type design structure is particularly adaptable for Shuttle payloads comprised of free flying satellites which will be deployed or retrieved in low Earth orbits.

3. This type of piggy back payload can be developed and operated either as a part of a planned Shuttle payload, or be used on a quick reaction, space available basis.

4. Impact of this type of payload on Shuttle and the primary payload is minimal. No crew involvement is required, subsystem support such as power distribution, data management, and thermal control is largely self-contained, and payload c.g. is generally unaffected to any significant degree (what affect there is often is positive).

5. Operation of the multi-mission payload is usually conducted on a non-interfering basis with primary payload operations such as deployments and retrievals. An advantage of this type of operation is that in general the full resources of Shuttle are available to the multi mission payload when it is operating.

6. Many types of experiment and sensor combinations are compatible with the essentially autonomous support structure characteristic of this type of payload.

7. Significant advantages of this type of payload are simplified integration and increased flight opportunities.

8. The STR subsystem requirements are compatible with NASA standard subsystem, i.e., computers, C&DH, etc.

Sensor Modifications

1. A majority of the sensors associated with the EVAL payloads studied to date have been/are being developed for applications on platforms other than Shuttle, i.e., free flying satellites, sounding rockets, balloons, and aircraft.

2. In general, sensors which have not been specifically designed for Shuttle applications will require modification. These modifications will involve changes for protection against the environment associated with Shuttle payloads, as well as design changes resulting from a different operating altitude and velocity, i.e., different resolution, swath widths, framing times, etc.

3. Those sensors presently existing in a hardware state can usually be suitably modified at a cost of several hundred thousand dollars and within an elapsed time frame of approximately one year.

4. Modification of sensors being developed for use with carriers other than Shuttle, but not presently existing, should be considered in two different scenarios. If their present development continues, and they are actually produced, then the conclusions of (3) above is appropriate. However, if the sensor development is terminated, then the adaptation of the sensor design for Shuttle applications can escalate the time and money involved to values on the order of 3 years and 5 to 10 million dollars. This results from the fact that not only must the sensor be redesigned, but it must also be fabricated, tested, and integrated.

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